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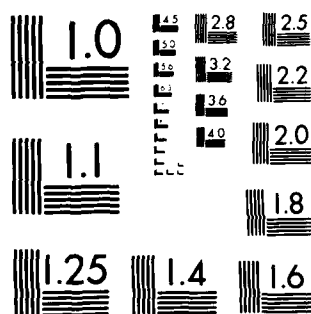
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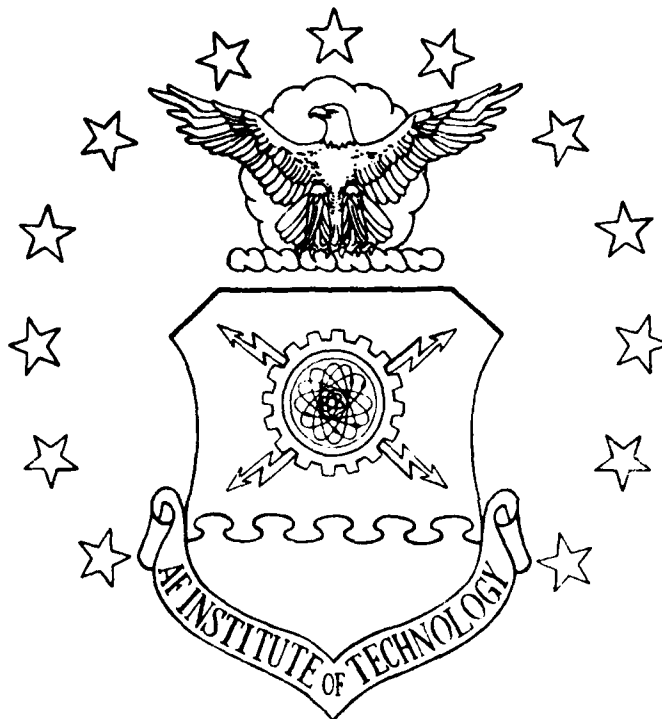
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A METHODOLOGY FOR SELECTION OF A
SATELLITE SERVICING ARCHITECTURE

VOLUME III, APPENDICES

DESIGN STUDY

AFIT/GSE/85D

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Wright-Patterson Air Force Base, Ohio

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A METHODOLOGY
FOR
SELECTION OF A SATELLITE SERVICING ARCHITECTURE
VOLUME III, APPENDICES
DESIGN STUDY

Presented to the Faculty of the School of Engineering
of the Air Force Institute of Technology

Air University

In Partial Fulfillment of the
Requirements for the Degree of
Master of Science

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Graduate Systems Engineering

December 1985

Approved for public release; distribution unlimited

Preface

The following report documents the design study of the Air Force Institute of Technology Graduate Systems Engineering Class of 1985. The report is in three volumes. The Executive Summary (Volume I) is a cursory review of the study and is meant to be self-contained. The Final Report (Volume II) and the Appendices (Volume III) are more detailed and should be read together for completeness. This study explains a two-phase methodology we developed to permit selection of an optimal military satellite servicing system. The work was conducted from December 1984 to December 1985. The original project concept and follow-on technical support was provided by the Rocket Propulsion Laboratory at Edwards AFB, California. Additional technical support and funding was provided by the Office for Manned Spaceflight (SD/YM) and the Office of Plans (SD/XR) at USAF Space Division, Los Angeles Air Force Station, California.

The faculty committee who assisted in this effort are:

Captain Stuart Kramer, Chairman

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Their help in reading and helping us revise countless draft copies of this work is greatly appreciated.



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A special measure of gratitude belongs to Major Dennis Clark for his help with the optimization program, and to Major Ken Feldman for his assistance with the value system. Our heartfelt thanks also goes to Mary Peltzer and Maggie Anderson, for their assistance with revisions of this document during the final hours.

We would also like to thank the following people for their assistance and guidance with different parts of this work. Without their help, parts of this effort would not have been possible: Major Don Brown, Mr. Robert Carlton, Colonel W.H. Crabtree, Colonel Gaylord Green, Colonel Donald G. Hard, Major James K. Hodge, Lt Colonel Janson, Mr. George Lemon, Lt Colonel Eric Sundberg, Lt Colonel Joseph Widhalm, Major G. V. Wimberly, Colonel William Wittress, and Colonel William F.H. Zersen.

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List of Symbols

Symbol	Description
Cr	Constraint Technique Constant
F	Feasible Region
Kg	Kilogram
Lu	Vector of Lower bounds of Control Variables
Lx	Vector of Lower bounds of State Variables
M	Maximal Eigenvalue
Mi	Initial total OSV mass
Mo	Final total OSV mass
Ms	Mass of OSV structure
P	Set of Elements Used in ISM
R	Relation
Sx	Feasible region in state space
Sz	Mapping feasible region into objective space
TOF	Time of flight for OSV mission
Ts	Synodic period
Tso	Period of the service orbit
Ttr	Period of the transfer orbit
Two	Period of the waiting orbit
U	Vector of Exogenous Variable
Ui	Exogenous Variable
Uu	Vector of Upper bounds on Control Variables
Ux	Vector of Upper bounds of State Variables

List of Symbols (Continued)

Symbol	Description
$V()$	Value Function
V_1	Velocity vector 1
V_2	Velocity vector 2
V_{po}	Velocity of parking orbit
V_{tra}	Velocity at apogee of transfer orbit
V_{trp}	Velocity at perigee of transfer orbit
V_{woa}	Velocity at apogee of waiting orbit
w_r	Weighted Technique Weight
X	Vector Of State Variables
X_i	i th State Variable
Z	Objective Function
Z	Vector of Performance Indices
Z_i	i th Performance Index
ΔV_a	Delta Velocity required at perigee
ΔV_{osv}	Delta Velocity reequred for OSV mission
ΔV_p	Delta Velocity required at perigee
$\Delta V_{resupply}$	Delta velocity required to travel from parking orbit to service orbit. enter service orbit. and return to parking orbit
ΔV_{wo}	Delta velocity required to get into and back out of waiting orbit
g_e	Gravitational accelleration at surface of earth
p_i	Element of Set P

List of Symbols (Continued)

Symbol	Description
ra	Radius at apogee of elliptical orbit
rearth	Normal radius of the earth
rp	Radius at perigee of elliptical orbit
rpo	Radius of parking orbit
rso	Radius of service orbit
tavgphase	Average time between OSV launch opportunities
tintersat	Travel time between satellites being serviced
tmaxphase	Maximum Time between OSV launch opportunities
tsatserv	Total time to service Y satellites (tservicing + tintersat)
tservicing	Time to service Y satellites

List of Abbreviations

Abbreviation	Description
AF	Air Force
CDC Cyber	Control Data Corp Cyber 175 Computer
DELIV	Delivered
DM	Decision Maker
DoD	Department of Defense
EMADAM	Extended Multi-Attribute Decision Analysis Model
EVA	Extravehicular Activity
FHG	Fixed High-G launch vehicle
FLG	Fixed Low-G launch vehicle
HLLV	Heavy Lift Launch Vehicle
HR	Hour
IC	Initial Cost
ICW	Initial Cost Weighting
ISM	Interpretive Structural Modeling
Isp	Specific Impulse of Fuel
KG	Kilogram
KTCN	Kuhn-Tucker Conditions for Noninteriority
Kg/Hr	Kilo Grams per Hour
Km	Kilometer
LEO	Low Earth Orbit
LG	Low-G launch vehicle
MADAM	Multi-Attribute Decision Analysis Model
MAUT	Multi Attribute Utility Theory

List of Abbreviations (Continued)

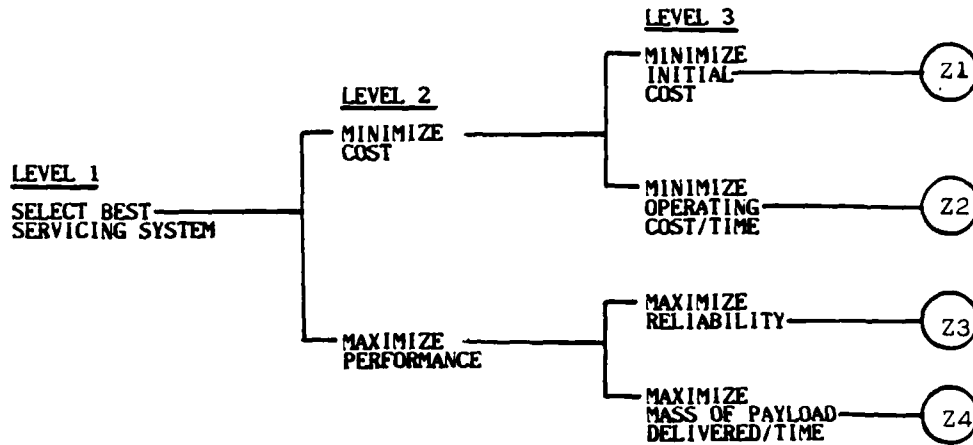
Abbreviation	Description
MAX	Maximize
MHG	Mobile High-G launch vehicle
MLG	Mobile Low-G launch vehicle
MMS	Multimission Modular Spacecraft
MOOT	Multiple Objective Optimization Theory
MPD	Mass of Payload Delivered
MPDW	Mass of Payload Delivered Weighting
MPI	Mutually Perferentially Independent
MWDI	Mutual Weak Difference Independent
NASA	National Aeronautics and Space Administraion
NDSS	Non-Dominated Solution Set
ASSDD-42	National Security Decision Directive 42
OC	Operating Cost
OCW	Overall Cost Weighting
OMV	Orbital Maneuvering Vehicle
OPCW	Operational Cost Weighting
OPS	Operating
OPW	Overall Performance Weighting
OSV	Ordital Servicing Vehicle
OTV	Orbital Transfer Vehicle
P/L	Payload
PERF	Performance

List of Abbreviations (Continued)

Abbreviation	Description
PI	Performance Index
PI	Performance Indices
PPI	Pairwise Preferentially Independent
PROCES	Computer Program for Vector Optimization Problems
PrI	Preferentially Independent
R&D	Research and Development
REL	Reliability
RFP	Request For Proposal
RMS	Remote Manipulator System
RW	Reliability Weighting
SB	Space Base
SE	Systems Engineering
SSS	Satellite Servicing System
SUMT	Sequential Unconstrained Minimization Technique
SV	State Variable
SYS	System
TAV	Transatmospheric Vehicle
TC	Total Cost
USAF	United States Air Force
VOP	Vector Optimization Problem
WDI	Weak Difference Independent

Appendix A

Satisfaction of MPI for Hierarchy of Objectives



Attributes: Z1 - Initial Cost
Z2 - (Operating Cost)/Time
Z3 - Reliability —
Z4 - (Mass of Payload Delivered)/Time

***** Hierarchy Tree of Objectives *****

Satisfaction of mutual preferential independence (MPI) among the attributes of an objective tree is a necessary and sufficient condition for using an additive value function (Feldman and Rowell, 1985). When three or more attributes are involved, satisfying pairwise preferential independence (PPI) is equivalent to satisfying MPI (Keeney and Raiffa, 1976:114). PPI and MPI are defined in Section 3.1.2.2. Because the objective tree above has four attributes, and

since satisfying PPI requires fewer steps, satisfaction of PPI (which, in turn satisfies MPI) is demonstrated below.

In order to satisfy PPI, the analyst or decision maker must test all possible combinations of pairs of attributes for at least two levels of their complementary sets.

This requires

$$2 * \frac{n!}{2!(n-2)!} \quad (A.1)$$

tests (Helstrom, 1984:35). In this case, $n=4$, and a total of twelve tests are required.

Each attribute in the complementary set of attributes should be set at a given level while the preferences between the attribute pairs are examined. The level of the complementary set should then be changed and the preference between the same pair of attributes should then be re-examined. If the preference did not change for that attribute pair when its complementary set changed values, that pair of attributes can be considered PPI of its complementary set.

It is recommended that the two levels used for the complementary sets be the least acceptable and the most acceptable values for those attributes. This test is, of course, a subjective one. One must be convinced that he can satisfy PPI in this case to justify using a linear value function.

The tests for the attributes in the above tree are below. where "IC" is initial cost, "OC" is (operating cost)/time, "R" is reliability, and "MPD" is (mass of payload delivered to orbit)/time.

ATTRIBUTE PAIR	COMPLEMENTARY SET AND LEVEL	PREFERENCES CHANGES?
{IC,OC}	{R,MPD}: lowest values	NO
{IC,OC}	{R,MPD}: highest values	NO
{IC,R}	{OC,MPD}: lowest values	NO
{IC,R}	{OC,MPD}: highest values	NO
{IC,MPD}	{OC,R}: lowest values	NO
{IC,MPD}	{OC,R}: highest values	NO
{OC,R}	{IC,MPD}: lowest values	NO
{OC,R}	{IC,MPD}: highest values	NO
{OC,MPD}	{IC,R}: lowest values	NO
{OC,MPD}	{IC,R}: highest values	NO
{R,MPD}	{IC,OC}: lowest values	NO
{R,MPD}	{IC,OC}: highest values	NO

Therefore, PPI and MPI are satisfied for the attributes in this hierarchy tree, and a linear value function is justified.

Appendix B

Detailed Heirarchy of Objectives for Selection of a Satellite Servicing System (SSS)

After discussions with decision makers at the USAF Space Division (Green, 1985; Lemon, 1985; Sundberg, 1985; Wimberly, 1985; Wittress, 1985; Zerson, 1985) a comprehensive hierarchy of objectives was developed, consisting of six levels and 40 objectives and subobjectives. Figure B.1 shows this hierarchy with labeled objectives and attributes. The first digit of the numeric label indicates the level of the objective within the hierarchy, and the second digit is used to identify each objective within that level. Attributes are found in the circles, and are identified by a number for each attribute, preceeded with the letter "A". The following is a description of each objective and attribute within the hierarchy.

Descriptions of Objectives Within Figure B.1

- 1-1 Military Satellite Servicing System Selection: To select the satellite servicing system which best meets the objectives presented within the hierarchy structure.
- 2-1 Satisfies Congressional Concerns: The system is desired to meet all requirements necessary that would impact Congress's criteria for selecting a system.
- 2-2 Mission Accomplishment: The system is desired to accomplish its mission by meeting various specifications. These specifications indicate how well a system could accomplish its mission.

- 2-3 Utilization of Limited Resources: This objective is to minimize the use of limited resources by a system. The limited resources are such things as energy, manpower, money, materials, and transportation infrastructure.
- 3-1 Internationally Politically Stabilizing: The system is not desired to internationally destabilize relations that exist between governments. Such things as a country's public safety, threat of attack, or breaking of treaties by use of a system could cause political destabilization of international relations.
- 3-2 Technological Risk: This objective is to minimize risks associated with selecting systems requiring new, advanced technology developments not already available.
- 3-3 National Prestige: The system considered is desired to favorably impact the view held of the this country by others.
- 3-4 Cost Versus Performance: The system is desired to meet its performance requirements for a reasonable cost.
- 3-5 Environmental Impact: The system should not adversely impact the environment.
- 3-6 Deployable Within Constraints: The system should be able to be implemented within apriori time requirements.
- 3-7 Can Supply Satellite Needs: It is desired that the system considered be able to meet satellite supply requirements.
- 3-8 Flexibility: The system is desired to be flexible, in that it is able to operate under various conditions. The system should handle anomalies during a mission, and be able to respond to anomalies by priority.
- 3-9 Survivability: How well can the system operate under man-induced, hostile environment. This objective is to operate "well" under hostile environment.
- 3-10 Availability: Is the system available based on its reliability and maintainability characteristics? The system is desired to be available over as much time as is possible.

- 3-11 Transportation Infrastructure: The system being considered is desired to be able to use existing transportation structures. A system requiring new roads, canals, ships, airplanes, to transport any components would not be as preferred as a system not requiring these.
- 3-12 Energy: This objective seeks to minimize energy use.
- 3-13 Money: This objective seeks to minimize system cost.
- 3-14 Scarce Strategic Materials: The system under consideration is desired to use a minimum amount of scarce strategic materials such as titanium, nuclear elements, or chrome.
- 4-1 Degree of Technology: To what degree does the system use new technology. Does the system utilize advanced concepts and materials.
- 4-2 Perception of System Reliability: The system is desired to be perceived as being very reliable in its performance of activities.
- 4-3 Americans in Space: The system is preferred to use increased numbers of Americans in space to perform its activities.
- 4-4 National Resource Consumption: The system under consideration is desired to use minimal amounts of the countries national resources.
- 4-5 Pollution: This objective is to minimize pollution created by a system. Such things as water, air, and land pollution are desired to be minimized in a system selection.
- 4-6 Public Safety: The system being considered is desired to not harm the public in its operation.
- 4-7 Handle Anomalies on Mission: How well can the system under consideration respond to unplanned events while performing a mission? The more a system is able to respond to such unplanned events would increase its possible selection.
- 4-8 Respond to Anomalies By Priority: How well can the system respond to unplanned events when that system is not performing a mission. Greater response capability would indicate a more preferred system.

- 4-9 Reliable: The system is desired to be reliable in its operation.
- 4-10 Maintainable: The system is desired to be maintainable to keep it operating.
- 4-11 System Life Cycle Costs: This objective is to minimize the life cycle costs incurred by the system being considered.
- 5-1 Spare Availability: Does the system have spare parts available to repair damaged equipment when the system is on-mission? A more preferred system would have such spare parts available.
- 5-2 Robotic Level: The system is desired to have sophisticated robotic technology incorporated in its design.
- 5-3 # of People On-board OSV: The system being considered is desired to have people on board to handle unplanned events occurring on-mission.
- 5-4 Spares Availability: Does the system have spare parts available to repair damaged equipment when the system is on-mission? A more preferred system would have such spare parts available.
- 5-6 Response Time: The system is desired to be able to respond quickly to anomalies by priority.
- 5-7 Operations Cost Per Time: This objective is to minimize operations costs per time for a system.
- 5-8 Amortized Initial Cost/Time: This objective is to maximize the system life, while minimizing the initial cost.
- 6-1 On-orbit Storage Capacity: The system is desired to have storage capacity to carry available spare parts.
- 6-2 R&D Cost Per System Life: This objective is to minimize the research and development costs in developing a system with a long life. The system is desired to have minimal research and development costs, and a "long" life.
- 6-3 Production Cost/Hardware Life: The system being considered is desired to have small production costs and a "long" hardware life.

Descriptions of the Attributes Within Figure B.1

- A1 Agrees With Policies and Treaties: Does the system agree with international policies and treaties in its operation?
- A2 Man On-board: Is the system manned or unmanned when operating in space?
- A3 R&D Invested: How much research and development is required for the system to be developed?
- A4 Net Savings Due: The measure of savings from servicing the satellite with the system, instead of replacing the satellite.
- A5 Time From IOC to FOC: The difference in time from Initial Operational Capability (IOC) of the system, till Full Operational Capability (FOC) of the system.
- A6 % Satellite Needs Met Per Time: The percent of the required satellite needs that can be met by the system.
- A7 Scenario: Description of an environment that the system is operating in to aid in measuring survivability. The system could be operating in a scenario of a peacetime environment, a nuclear war environment, or a conventional war environment.
- A8 R&D Invested: How much research and development is required for the system to be developed?
- A9 Fuel Specific Impulse: The specific impulse of the fuel for various components of the system requiring fuel.
- A10 Structural Mass Ratio: The structural mass ratio for the various components of the system.
- A11 OSV Spare Parts Storage: The OSV spare parts storage capacity for a system that uses OSVs.
- A12 Robotic R&D Costs: The OSV costs incurred for research and development in automating the operation of the OSV for a system.
- A13 # Robots On-board: The number of robots required on-board an OSV in the performance of its mission.

A14 Scenario: A description of the environment within which the system would be operating. Some examples are a peacetime environment, nuclear war environment, or a conventional war environment.

A15 Depot Storage Capacity: The capacity within a system to store necessary spares, in space, at a central location.

A16 OSV Payload Capacity: The OSV payload capacity which could be used to carry necessary spare parts.

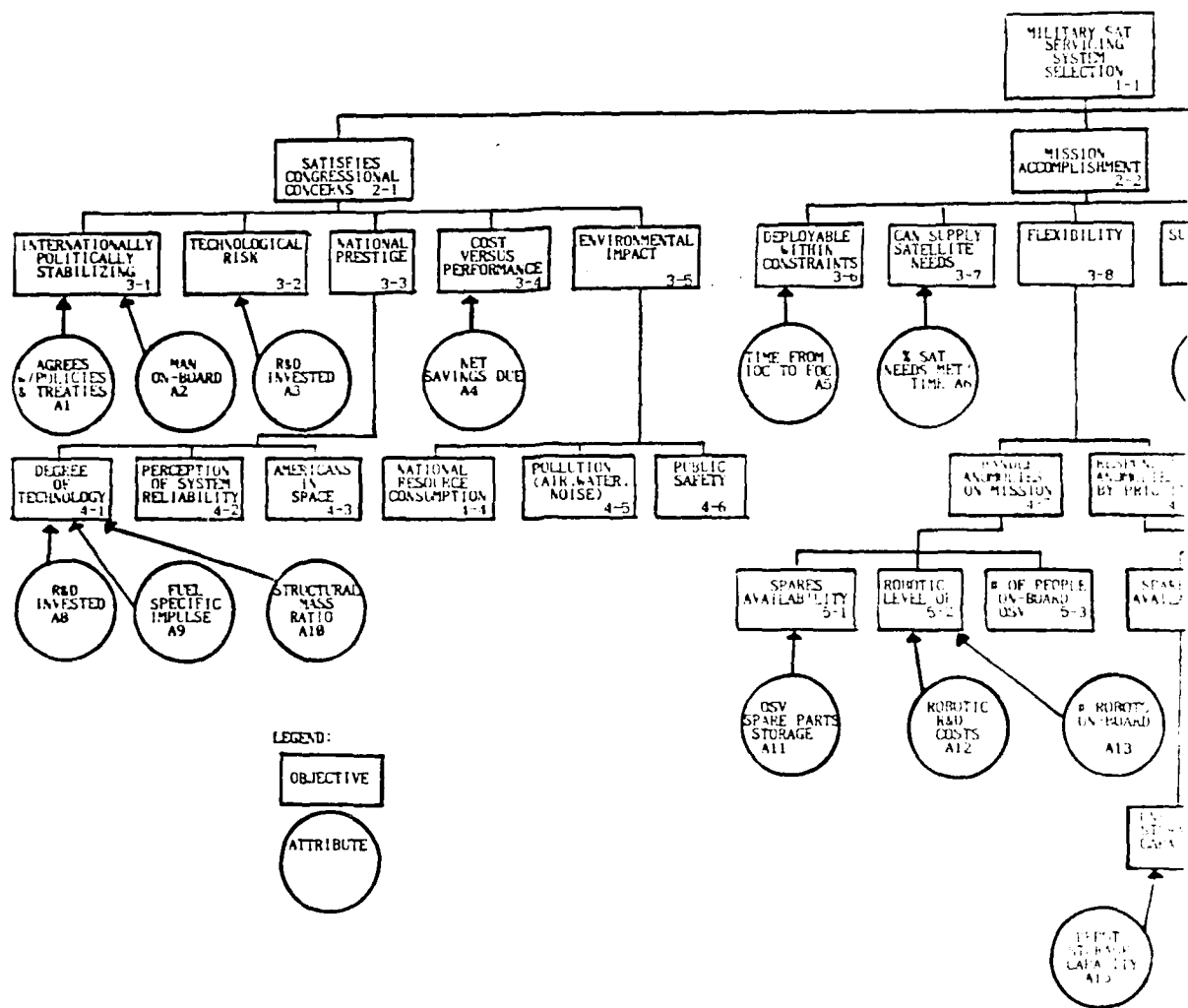
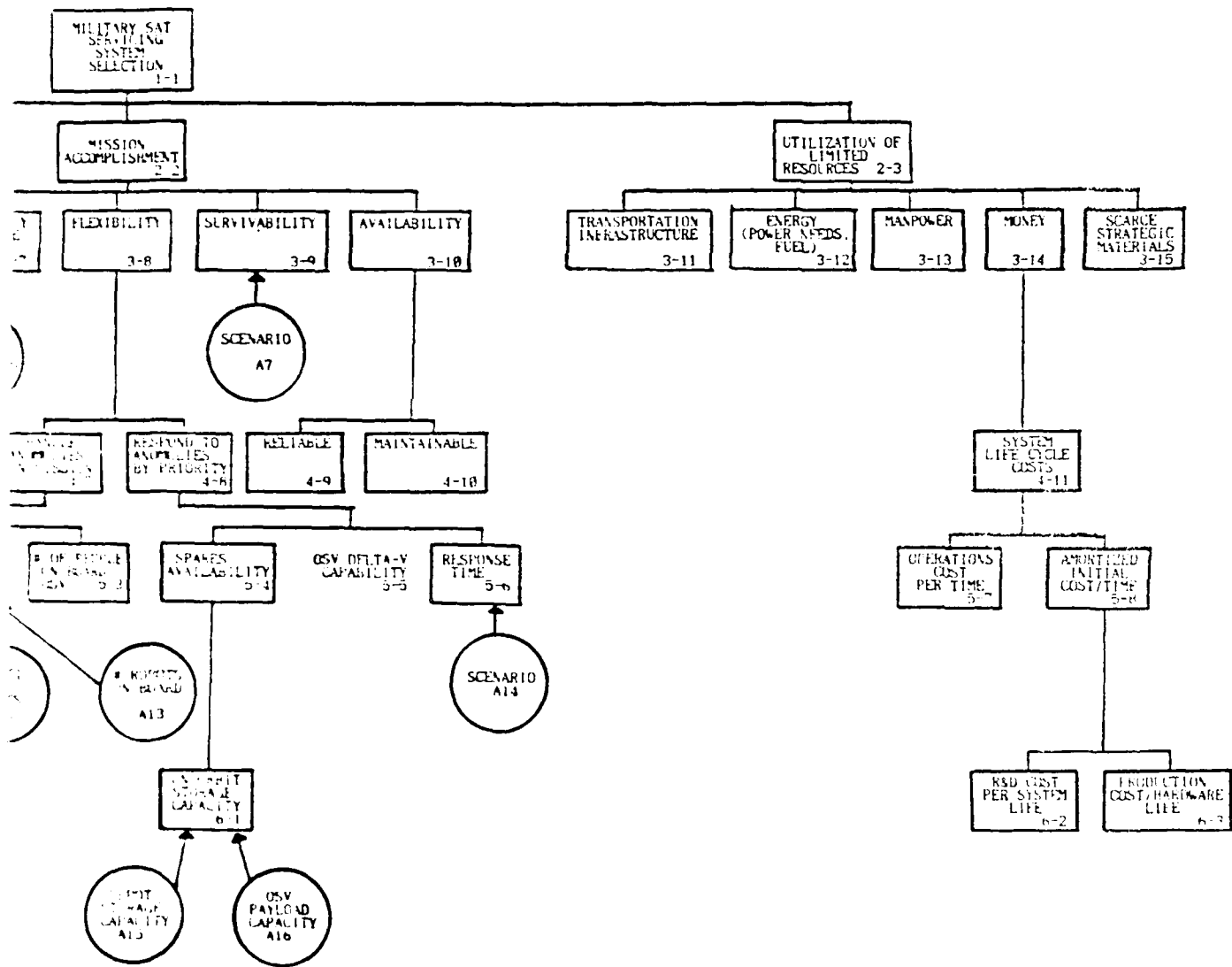


Figure B.1 Comprehensive Hierarchy

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Comprehensive Hierarchy of Objectives

Appendix C

Computer Program for Calculation of Figures of Merit Using Linear Value Functions

```
PROGRAM FOM
C PROGRAM TO CALCULATE FIGURE OF MERIT FROM NDSS
C
C VARIABLE DECLARATION
  INTEGER NS,NP
  REAL P(100,100),PMA(10),PMIN(10),V(100,100),OP,TC,IC,OC,R,MPD
  REAL INFOM1,INFOM2,TFOM(100)
C IDENTIFY FILES FOR INPUT NDSS AND OUTPUT DATA
  OPEN (UNIT=10,FILE='ndss')
  OPEN (UNIT= 3,FILE='data')
  OPEN (UNIT= 4,FILE='rank')
  OPEN (UNIT= 7,FILE='val')
  REWIND 10
  REWIND 7
C
C READ IN # OF NONDOMINATED SOLUTIONS AND # OF PERFORMANCE INDICIES
  NS=69
  NP=4
C
C READ IN THE PERFORMANCE INDEX MEASURES FOR EACH SYSTEM REPRESENTATION
  DO 50 J=1,NS
    READ (10,1000) (P(J,I),I=1,NP)
  1000 FORMAT (4E17.9)
  50 CONTINUE
C
C WRITE THE PERFORMANCE INDEX MEASURES READ IN, AND THE SYSTEM #
  WRITE (3,1100)
  1100 FORMAT ('SYSTEM #',9X,'OPERATING ',9X,'INITIAL ',8X,
    *'RELIABILITY',7X,'MASS OF ')
  WRITE (3,1101)
  1101 FORMAT (20X,'COST',13X,'COST',23X,'OF PAYLOAD')
  WRITE (3,1102)
  1102 FORMAT (69X,'DELIVERED')
  WRITE (3,1105)
  1105 FORMAT (17X,'DOLLARS/HR',9X,'DOLLARS',11X,'PERCENT',13X,'KG')
  WRITE (3,*) ''
  DO 60 J=1,NS
    WRITE (3,1110) J,(P(J,I),I=1,NP)
  1110 FORMAT (2X,13,7X,4E17.9)
  60 CONTINUE
C
C FIND THE MAX AND MIN VALUE OF EACH PERFORMANCE INDEX
  DO 70 I=1,NP
    DO 80 J=1,NS
      PMA(I) = AMAX1(PMA(I),P(J,I))
      PMIN(I) = AMIN1(PMIN(I),P(J,I))
    80 CONTINUE
```

```

70 CONTINUE
C
C DETERMINE LINEAR VALUE FUNCTION VALUE FOR EACH PERFORMANCE INDEX
C IN EVERY SYSTEM
  DO 100 J=1,NS
    DO 90 I=1,2
C    THIS FUNCTION GIVES VALUES THAT MINIMIZE COSTS
      V(J,I) = (PMAX(I) - P(J,I)) / (PMAX(I) - PMIN(I))
    90 CONTINUE
C
C THIS FUNCTION GIVES VALUES THAT MAXIMIZE RELIABILITY AND MASS
C OF PAYLOAD DELIVERED
  DO 95 I=3,4
    V(J,I) = (P(J,I) - PMIN(I)) / (PMAX(I) - PMIN(I))
  95 CONTINUE
100 CONTINUE
C
C WRITE VALUE FUNCTION VALUES FOR EACH PI IN EVERY SYSTEM
  WRITE (3,*) ''
  WRITE (3,*) 'LINEAR VALUE FUNCTION VALUES'
  WRITE (3,*) ''
  WRITE (3,1100)
  WRITE (3,1101)
  WRITE (3,1102)
  WRITE (3,*) ''
  DO 110 J=1,NS
    WRITE (3,1210) J,(V(J,I),I=1,NP)
    WRITE (7,1211) (V(J,I),I=1,NP),J
1210 FORMAT (2X,13,7X,4E17.9)
1211 FORMAT (4E13.7,3X,13)
  110 CONTINUE
C
C READ IN WEIGHTINGS FOR HIERARCHY OF OBJECTIVES FROM DECISION MAKER
C PREFERENCES
  WRITE (*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR',
  *'OVERALL PERFORMANCE?'
  READ (*,*) OP
  WRITE (*,*) 'OVERALL PERFORMANCE WEIGHTING IS',OP
  WRITE (4,*) 'OVERALL PERFORMANCE WEIGHTING IS',OP
  WRITE (*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR TOTAL COST?'
  READ (*,*) TC
  WRITE (*,*) 'TOTAL COST WEIGHTING IS',TC
  WRITE (4,*) 'TOTAL COST WEIGHTING IS',TC
  WRITE (*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR INITIAL COST?'
  READ (*,*) IC
  WRITE (*,*) 'INITIAL COST WEIGHTING IS',IC
  WRITE (4,*) 'INITIAL COST WEIGHTING IS',IC
  WRITE (*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR',
  *'OPERATING COST/HR?'
  READ (*,*) OC
  WRITE (*,*) 'OPERATING COST/HR WEIGHTING IS',OC
  WRITE (4,*) 'OPERATING COST/HR WEIGHTING IS',OC
  WRITE (*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR RELIABILITY?'

```

```

      READ (.*,*) R
      WRITE (.*,*) 'RELIABILITY WEIGHTING IS',R
      WRITE (4,*) 'RELIABILITY WEIGHTING IS',R
      WRITE (.*,*) 'WHAT IS THE HIERARCHY WEIGHTING FOR'.
      *'MASS OF PAYLOAD DELIVERED?'
      READ (.*,*) MPD
      WRITE (.*,*) 'MASS OF PAYLOAD DELIVERED WEIGHTING IS',MPD
      WRITE (4,*) 'MASS OF PAYLOAD DELIVERED WEIGHTING IS',MPD
C
C  CALCULATE THE FIGURE OF MERIT FOR EACH SYSTEM REPRESENTATION
C  USING THE WEIGHTINGS AND VALUES APPLIED TO THE HIERARCHY TREE
      DO 130 I=1,NS
      INFOM1=0.0
      INFOM2=0.0
      INFOM1=((OC*V(I,1))+(IC*V(I,2)))*TC
      INFOM2=((R*V(I,3))+(MPD*V(I,4)))*OP
      TFOM(I)=INFOM1+INFOM2
C
C  WRITE THE FIGURE OF MERIT FOR EACH SYSTEM
      WRITE (3,*) ''
      WRITE (3,*) TFOM(I), 'IS THE FIGURE OF MERIT FOR SYSTEM',I
      WRITE (4,*) TFOM(I), 'IS THE FIGURE OF MERIT FOR SYSTEM',I
130 CONTINUE
      END

```

Appendix D

Model Equations

This appendix presents the equations used to model the LG+OSV, FHG+LG+OSV, and SB+FHG+LG+OSV Satellite Servicing Systems. The Performance Index equations are in section D.1 and the State and Constraint equations common to the three models are in section D.2. Additional equations added to this common set for the LG+OSV, FHG+LG+OSV, and SB+FHG+LG+OSV models are in sections D.4, D.5, and D.6 respectively. A complete descriptive listing of all variables (state, intermediate, and exogenous) is in Appendix E and the intermediate equations are in Appendix F.

D.1 Performance Index Equations

Z1 is Operations Costs

Equation D.1

$$Z1 = I04 + I45 + I05 + I46$$

I04 = Launch Site Operations Cost

$$= 40 * (U79 / 8760) * (I03 ** 0.34) / [(1 / X335) ** 0.55] \\ * (X320) / X335$$

I05 = LG Fuel Cost

$$= X326 * U31 * X301$$

I46 = OSV Manned Cost

$$= X500 * X555 * U78$$

I45 = OSV Fuel Cost

$$= X526 * U22 * X501$$

Z2 is Initial Costs

Equation D.2

$$Z2 = I07 + I06 + I47 + I48$$

I06 = LG R&D Cost

$$= 6500 * U79 * (X325 ** 0.21) * U37 * U38 * [EXP(2 * X345)]$$

I07 = LG Production Cost

$$= X300 * 12 * (X325 ** 0.56) * \{X300 ** [LN(U32) / LN(2.0)]\} \\ * U79 * [EXP(2 * X345)]$$

I47 = OSV R&D Cost

$$= 6500 * U79 * (I44 ** 0.21) * U29 * U30 * [(U21) ** X555] \\ * [EXP(2 * X545)]$$

I48 = OSV Production Cost

$$= X500 * (16.5) * (I44 ** 0.56) \\ * \{X500 ** [LN(U23) / LN(2.0)]\} * U79 * [EXP(2 * X545)]$$

Z3 is Reliability

Equation D.3.1

Reliability for the LG+OSV and FHG+LG+OSV models.

$$Z3 = \{1 - [(1 - X345) ** X300]\} * \{1 - [(1 - X545) ** X500]\}$$

Equation D.3.2

Reliability for the SB+FHG+LG+OSV model.

$$Z3 = \{1 - [(1 - X545) ** X500]\}$$

Z4 is Mass Delivered to the Satellites per Time (Hr)

Equation D.4

$$Z4 = X501 * X510$$

$$= (\#OSV \text{ missions/time}) * (OSV \text{ payload})$$

D.2 State and Constraint Equations Common to all Models

Equation D.5

Inequality equation placing upperbounds on the time between LG launches due to maximum time people can remain in space.

$$1/X301 \leq U77$$

(time between LG launches)

$$\leq (\text{max time people can be in space})$$

Equation D.6 and D.7

Two inequality equations placing bounds on the time between LG launches from a specific launch site.

$$X335 \rightarrow U48$$

$$U49 \rightarrow X335$$

Equation D.8

Inequality equation placing lowerbounds on the required time between LG launches for refurbishment of the LG vehicle. This equation assumes the launch vehicle must sit on the ground for a U40 time period.

$$X330 \rightarrow U40$$

Equation D.9

Equality equation relating the LG launch rate to the launch capability of the launch sites.

$$X301 = X320/X335$$

$$\#LG \text{ launches} =$$

$$(\# \text{ launch sites}) / (\text{time between launches from a site})$$

Equation D.10, D.11

Equations placing upper and lower bounds on the LG vehicle's structural mass ratio.

LG structural mass ratio $[I28 = X325/(X326+X325)]$ becomes

$$I28 \rightarrow U34$$

$$I28 \leftarrow U36$$

Equation D.12

Inequality equation requiring the LG to have one or more stages.

$$X370 \rightarrow 1$$

Equation D.13, D.14

Inequality equations placing upper and lower bounds on the LG reliability.

$$X345 \leftarrow 1$$

$$X345 \rightarrow 0$$

Equation D.15

Equality equation equating the delta velocity the LG must experience to attain orbit to the thrust the mlg must provide. Equations for LG delta velocity are from (Hill and Peterson, 1970).

$$U33*U93*X370*LN\{129/[(128*(129-1))+1]\} = 121$$

$$(V_{mlg}) = (I_{sp} \text{ LG fuel}) * (\text{gravitational constant})$$

$$* (\# \text{ LG stages}) * LN[(\text{LG gross lift off weight})$$

$$/(\text{empty LG mass} + \text{LG payload})]$$

$$129 = [(X325 + X326 + X310) / X310] ** (1 / X370)$$

$$121 = \text{SQRT}[U90 / (X360 + U91)] + 5486 + 1152 + 1097 + 0$$

$$128 = X325 / (X325 + X326)$$

Equation D.16

Inequality equation placing upper bounds on the OSV mission length due to the maximum time people can remain in space.

$$X500 / X501 <= U77$$

Equation D.17

Inequality equation requiring the OSV to have one or more waiting orbits.

$$X565 \rightarrow 1$$

Equation D.18, D.19

Equations placing upper and lower bounds on the OSV vehicle's structural mass ratio.

OSV structural mass ratio [I42 = X525/(X526+X525)]

becomes

I42 -> U12

I42 <- U13

Equation D.20, D.21

Inequality equations placing upper and lower bounds on the OSV reliability.

X545 <- 1

X545 >- 0

Equation D.22

Equality equation relating the OSV delta velocity for one mission to the OSV payload and required fuel for the mission.

I35 = I33*(X560-1)+I32

= U24*U93*LN{[I44+X526+X510]/[I44+X526]}

(Vosv required) = (Vosv is capable of providing)

I34 = (U1+U91)*[(U2*X565-1)/(U2*X565)]**(2./3.)

I33 = 2*ABS{SQRT[U90/(U1+U91)]

-SQRT[2*U90/(U1+U91)-U90/I34]}

I36 = SQRT{2*ABS[U90/(U1+U91)-U90/(X360+U1+2*U91)]}

I37 = SQRT{2*ABS[U90/(X360+U91)-U90/(X360+U1+2*U91)]}

I32 = 2*ABS{SQRT[U90/(U1+U91)]-I36+I37

-SQRT[U90/(X360+U91)]}

Equation D.23

Equality equation relating the OSV payload to the number of satellites serviced per mission and the mass delivered to each satellite.

$$X510 = X561 * X560$$

$$\text{OSV payload mass} =$$

$$(\text{avg mass delivered}) * (\# \text{ satellites serviced/mission})$$

Equation D.24

Equality equation relating the number of OSV missions per time to the number of OSV vehicles and time from the start of one mission until the start of the next mission.

$$X501 = X500 / (I51 + I50)$$

$$\# \text{OSV missions/time} =$$

$$(\# \text{OSV}) / (\text{time from one launch until the next})$$

$$I50 = 2 * U92 * \{ [(2 * U91 + U1 + X360) / 2.] ** 1.5 \} / \text{SQRT}(U90)$$

$$+ (X560 - 1) * \{ 2 * U92 * [(U1 + U91) ** 1.5] \}$$

$$* (U2 * X565 - 1.) / [U2 * \text{SQRT}(U90)] \} + X560 * I52$$

$$I51 = X510 / U25$$

Equation D.25

Equality equation relating the total payload mass carried by OSV/time to the total mass delivered to satellites/time.

$$(X501) * (X510) = (U2) * (X561) / U6$$

Equation D.26

Equality equation relating the number of satellites serviced by the OSV to the total number of satellites requiring servicing.

$$X501 * X560 = U2 / U6$$

$$(\#OSV \text{ Mission/time}) * (\#Sat \text{ serviced/OSV Mission}) = \\ (\text{total \#satellites}) / (\text{service interval})$$

Equation D.27

Inequality equation placing lowerbounds on the OSV-LG rendezvous altitude, FHG and LG target altitude, and SB location altitude.

$$X360 \rightarrow U58$$

D.3 LG+OSV Model

For the LG+OSV model the following equations must be added to the group of common equations.

Equation D.28

Equality equation requiring the LG to carry enough payload into space to satisfy OSV and satellite mass requirements.

$$(X301) * (X310) = [141 + X501 * X510]$$

$$(\#LG \text{ missions/time}) * (LG \text{ payload}) = (OSV \text{ needs}) \\ + (\text{satellite needs})$$

Equation D.29

Inequality equation requiring the OSV fleet to be able to hold the payload from one LG.

$$X500 * (X510 + X526 + U19 + X555 * (U76 + 100 / U77) / X501) \rightarrow X310$$
$$(\#OSV) * (OSV \text{ payload} + OSV \text{ needs}) \rightarrow LG \text{ payload}$$

Equation D.30

Equality equation relating the LG mission rate to the number of LGs and the time between launches of a single LG.

$$X301 = X300 / (I12 + X330)$$

$$(\#LG \text{ launches/time}) =$$

$$(\#LG's) / (\text{total time between launches of a LG})$$

$$I13 = \{ \text{SQRT} [U90 / (X360 + U91)] + 5486 + 1152 + 1097 \} / (2 * U93)$$

$$I14 = X310 / U25$$

$$I15 = 3.$$

$$I12 = I13 + I14 + I15$$

D.4 FHG+LG+OSV Model

For the FHG-LG-OMV model the following equations must be added to the group of common equations. LG+OSV equations D.29 and D.30 are also required in the FHG+LG+OSV model.

The following operations costs must be added to the common operating costs. Z1 becomes:

Equation D.31

$$Z1 = I04 + I05 + I45 + I46 + I106 + I115 + I116$$

I106 = Cost of FHG Ground Based Energy/Time

$$= X100*(2*I105)*U111$$

I115 = FHG Launch Site Ops Cost/Time

$$= X120*U112*(X135**2)$$

I116 = FHG Vehicle Costs/Time

$$= X100*\{(1.057*0.0624534*I103*I108) + (46330*[(0.4536*I113/9)**0.77]) + (U114*I113)\}$$

Equation D.32

The following initial costs must be added to the common initial costs. Z2 becomes:

$$Z2 = I07+I06+I47+I48+I118+I119+I120$$

I118 = FHG Vehicle R&D Costs

$$= \{[3.235*0.0624534*I103*I108] + [7414460+22600*0.4536*I113/9]\}$$

I119 = Launch Site R&D Costs (launcher and power source or plant)

$$= U117*X360*(I103**3)$$

I120 = Launch Site Purchase or Fabrication Cost

$$= X120*\{[U118*(I103**3)] + [U119*(2*I105)**0.5]\}$$

Equation D.33

Inequality equation for the minimum amount of mass the LG must carry into orbit. The right hand side of the inequality represents the mass that must be launched by a low-G launch system.

$$X301*X310 \rightarrow (U14*X561*U2/U6)+I40+(X555*X500*U76)$$

$$+(U19 \times X501)$$

(LG carried mass/time) \rightarrow (low-G satellite needs/time)

+ (OMV req people mass/time)

+ (OMV life support mass/time)

+ (OMV parts needs/time)

Equation D.34

Inequality equation placing a lower bound on the time between launches from a particular FHG launch site.

$$X135 \rightarrow U120$$

Equation D.35

Inequality equation placing lower bound on the payload mass of the FHG. The equation guards against the FHG launching ridiculously small payloads.

$$X110 \rightarrow U123$$

Equation D.36

Equality equation defining the relationship between the required FHG mission/launch rate and the required time between launches at a site and the number of launch sites.

$$X100 = X120/X135$$

#FHG launches/time = (#launch sites)

/time between launches from a single site

Equation D.37 (revision of equation D.28)

Equality equation defining the relationship between

the mass launched into space by the two launch systems and
the mass required in space by the satellites and the OSV.

$$\begin{aligned} & (X301)*(X310)+(X100)*(X110) = (I41+X501*X510) \\ & (\#LG \text{ Mission/time})*(LG \text{ Payload}) \\ & +(\#FHG \text{ Missions*Usable FHG Payload/Mission}) \\ & = (OMV \text{ needs/time})+(\text{mass delivered to satellites/time}) \end{aligned}$$

D.5 SB+FHG+LG+OSV Model

For the SB-FHG-LG-OMV model the following equations must be added to the group of common equations. FHG+LG+OSV equations D.34, E.35 and D.36 are also required in the SB+FHG+LG+OSV model.

Equation D.38 (revision of D.T and D.31)

The following operations costs must be added to the common operating costs. Z1 becomes:

$$Z1 = I04+I05+I45+I46+I106+I115+I116+I130+I131$$

$$I106 = \text{Cost of FHG Ground Based Energy/Time}$$

$$= X100*(2*I105)*U111$$

$$I115 = \text{FHG Launch Site Ops Cost/Time}$$

$$= X120*U112*(X135**-2)$$

$$I116 = \text{FHG Vehicle Costs/Time}$$

$$\begin{aligned} & = X100*\{(1.057*0.0624534*I103*I108) \\ & \quad + (46330*[(0.4536*I113/9)**0.77])+(U114*I113)\} \end{aligned}$$

$$I130 = \text{SB Manned Cost/Time}$$

$$= X400*X415*U78$$

$$\begin{aligned}
 I131 &= \text{SB Replenishment Costs/Time (for fuel+parts)} \\
 &= FHG5*U134 + I124*U139
 \end{aligned}$$

Equation D.39 (revision of equations D.2 and D.32)

The following initial costs must be added to the common initial costs. Z2 becomes:

$$Z2 = I48+I07+I06+I47+I118+I119+I120+I134+I135+I136$$

$$\begin{aligned}
 I118 &= \text{FHG Vehicle R\&D Costs} \\
 &= [(3.235*0.0624534*I103*I108) \\
 &\quad + (7414460+22600*0.4536*I113/9)]
 \end{aligned}$$

$$\begin{aligned}
 I119 &= \text{Launch Site R\&D} \\
 &= U117*X360*(I103**0.5)
 \end{aligned}$$

$$\begin{aligned}
 I120 &= \text{Launch Site Purchase or Fabrication Costs} \\
 &= X120*\{[U118*(I103**0.5)]+[U119*(2*I105)**0.5]\}
 \end{aligned}$$

$$\begin{aligned}
 I134 &= \text{SB R\&D Costs} \\
 &= U136*(X425**0.5)+U135*(X415**0.75)
 \end{aligned}$$

$$\begin{aligned}
 I135 &= \text{SB Production or Fabrication Costs} \\
 &= X400*U137*(X425**0.5)
 \end{aligned}$$

$$\begin{aligned}
 I136 &= \text{SB Development Costs} \\
 &= X400*U138*X425
 \end{aligned}$$

Equation D.40 (revision of equation D.29)

Inequality equation requiring a single SB to be capable of holding an LG entire payload and the required SB safety level.

$$X420 \rightarrow X310+[U130*(I41+X510*X501+I124+I125+I126+I127)]$$

Equation D.41 (revision of equation D.33)

Inequality equation for the minimum amount of mass the LG must carry into orbit. The right hand side of the inequality represents the mass that must be launched by a low-G launch system.

$$X301 * X310 \rightarrow (U14 * X561 * U2 / U6) + I40 + (X555 * X500 * U76) \\ + (U19 * X501) + I125 + I126 + I127$$

(LG carried mass/time) \rightarrow (low-G satellite needs/time)

+ (OSV required people mass/time)

+ (OSV life support mass/time)

+ (OSV parts needs/time) + (SB needs)

Equation D.42

Equality equation for the SB structure mass to SB mass storage capacity. This is a way of insuring the SB structure is large enough to hold the mass the SB is expected to store.

$$X425 / X420 = U133$$

Equation D.43 (revision of equations D.37 and D.28)

$$(X301) * (X310) + (X100) * (X110) = (I41 + X501 * X510) \\ + I124 + I125 + I126 + I127$$

#LG missions/time * (LG payload)

+ (#FHG missions * usable FHG payload/mission)

= (OMV needs) + (mass del to Sat/time) + (SB needs)

Equation D.44 (revision of equations D.30)

Equality equation relating the LG mission rate to the number of LGs and the time between launches of a single LG.

$$X301 = X300 / (I12 + X330)$$

(#LG launches/time)

$$= (\text{\#LG's}) / (\text{total time between launches of a LG})$$

$$I13 = \{ \text{SQRT} [U90 / (X360 + U91)] + 5486 + 1152 + 1097 \} / (2 * U93)$$

$$I14 = X310 / U35$$

$$I15 = 3.$$

$$I12 = I13 + I14 + I15$$

$$X301 = X300 / (I12 + X330)$$

(#LG launches/time)

$$= (\text{\#LG's}) / (\text{total time between launches of a LG})$$

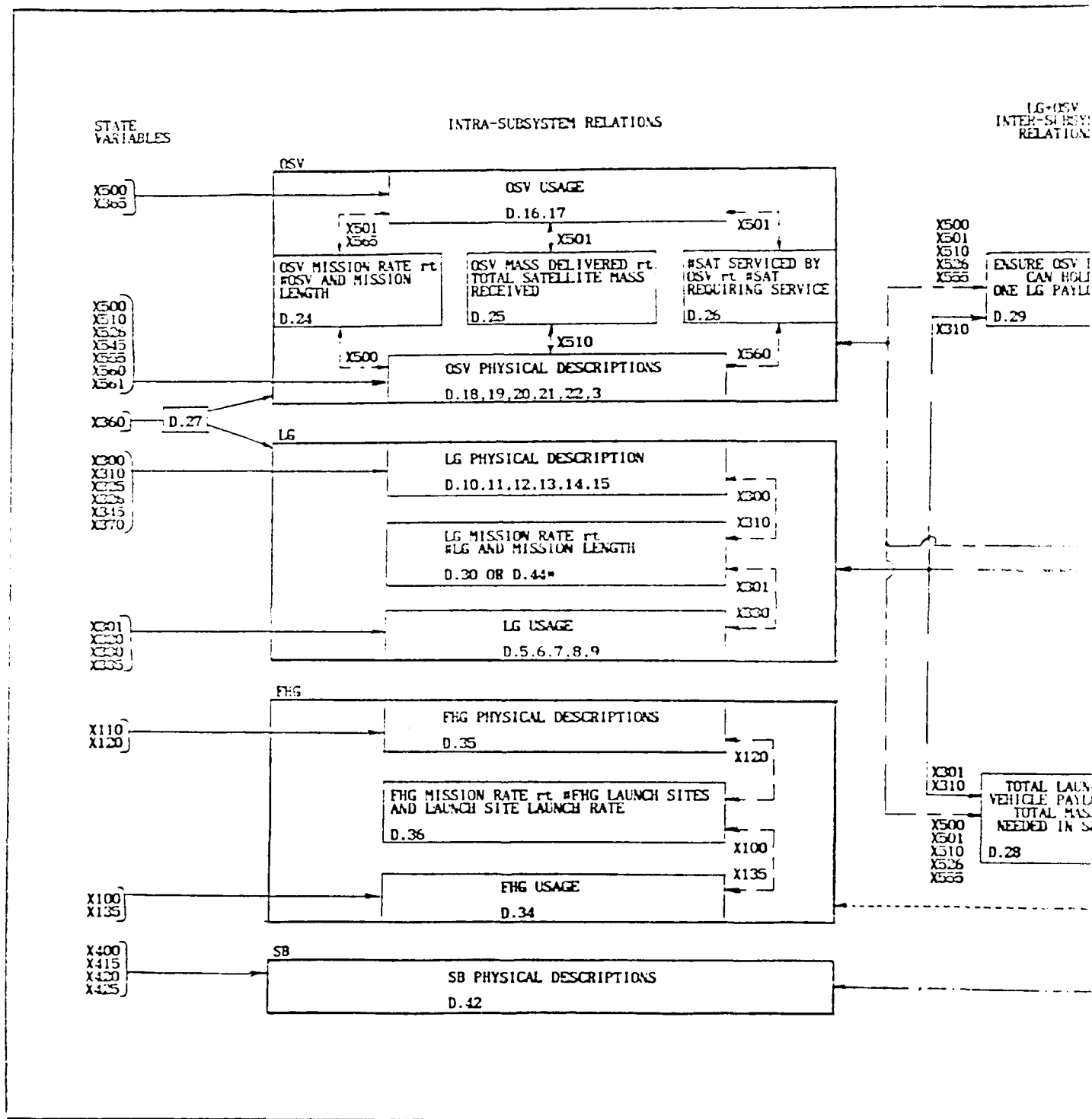
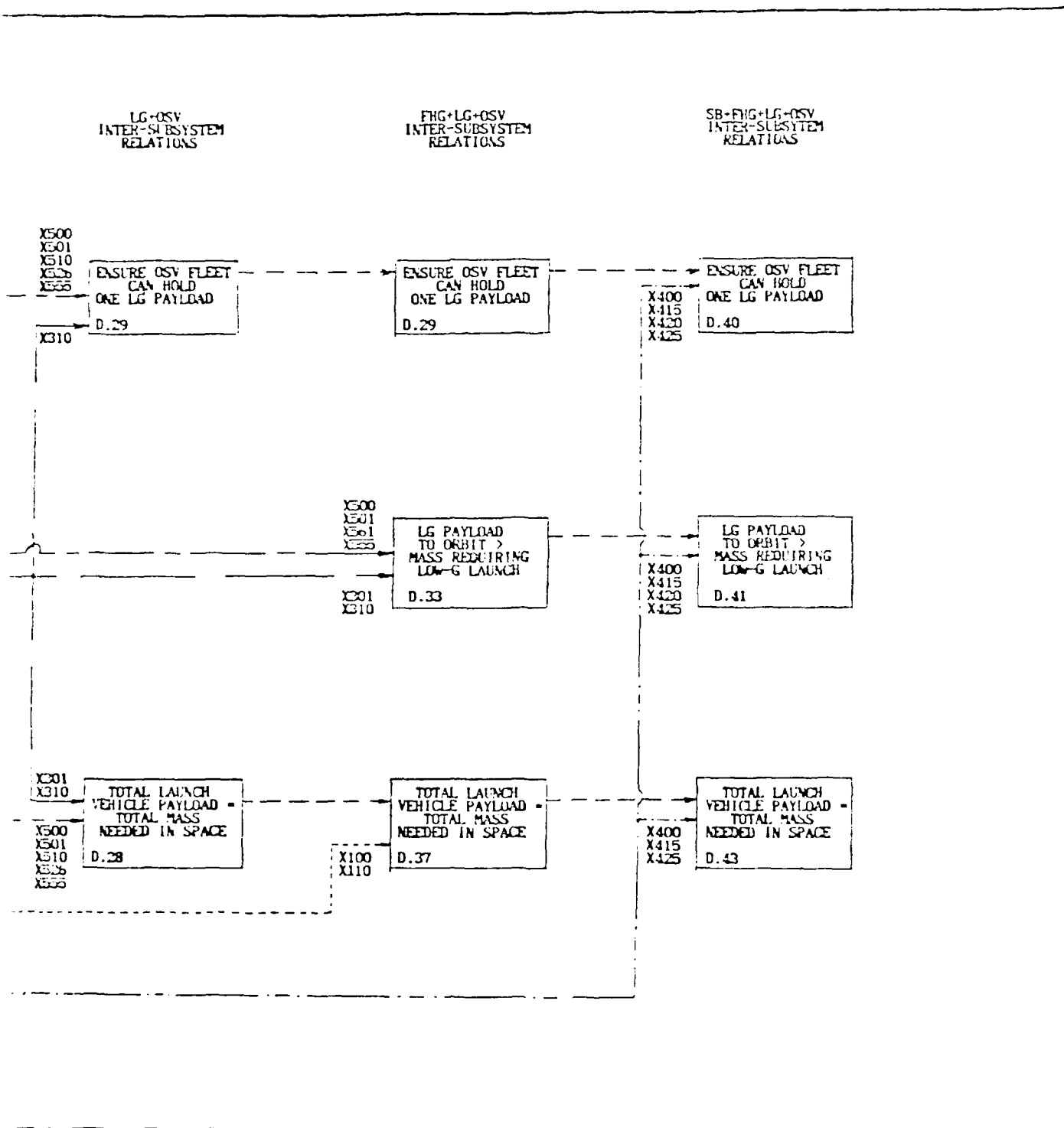


Figure D.1 Physical Model



Physical Model Schematic

Appendix E
Model Variables

Low-G State Variables

- X300 - # of LG systems
- X301 - # LG missions/time (mission rate/hr)
- X310 - Payload mass per launch [kg/launch]
- X320 - # of launch sites
- X325 - Empty vehicle structure mass [kg]
- X326 - Mass of propulsion fuel [kg]
- X330 - Average time between missions/LG [hrs]
(downtime for all reasons)
- X335 - Average time between launches from a specific
launch site [hrs]
- X345 - Reliability of LG [0<X345<1]
- X360 - OSV-LG rendezvous altitude
- X370 - # LG stages

OSV State Variables

- X500 - # of OSVs
- X501 - # OSV missions/time (OSV mission rate)
- X510 - Payload mass per OSV mission [kg/mission]
- X525 - Mass of OSV empty [kg]
(without guidance and life support equip)
- X526 - Mass of propulsion fuel req/mission[kg/mission]
- X545 - Reliability of an OSV [0<X545<1]
- X555 - OSV crew size (# of people)
- X560 - # satellites serviced/OSV mission [number/mission]

X561 - Average mass delivered to a satellite/service
(mass/service)

X565 - # of waiting orbits (n in the orbital equations)

Fixed High-G State Variables

X100 - # of FHG launches/time

X110 - Usable payload mass/launch
(mass usable by OSVs, SBs, satellites, etc)

X120 - # of FHG launch sites

X135 - Time between launches at a specific site [hrs]

SB State Variables

X400 - # of Space-bases (SB)

X415 - # of people to operate the SB

X420 - SB mass storage capacity

X425 - SB structure mass —

Exogenous Variables

The exogenous variables are input variables used to input constants into the model. When possible the exogenous variables are taken from some reliable source document or an expert in the area the exogenous is involved with. The variables that have as a source "group est" were either not found in the literature or were not in a form usable in the model. In these cases the study group came up with an estimate that seemed reasonable from an engineering point of view. As more work in the area of satellite servicing takes place in the future more accurate values for the exogenous

can easily be input and the model rerun to find the effect of the changes.

- U1 - Average satellite altitude [km]
200 < U1 < 23000 nominal = 800 kilometers
Source. group est
- U2 - # of satellites
12 < U2 < 500 nominal = 144 satellites
Source. group est
- U6 - Satellite service interval [hrs]
(time between services to the same satellite)
8760 < U6 < 43800 nominal = 26280[3 yrs]
Source. General Dynamics, 1983
- U12 OSV mass structure ratio lower bound
U12 = 0.1
Source. Hill and Petterson, 1970:328
- U13 OSV mass structure ratio upper bound
U12 = 1.0
Source. Hill and Petterson, 1970:328
- U14 - Percent of satellite required mass which
requires low-G launch
0.0 < U14 < 1.0 nominal = 0.25 or 25%
Source. group est.
- U16 - Coef for computation of OSV life support equip[kg]
2432 = U16
Source. Bolen, 1980:Ch3

U17 - Coef for computation of OSV life support equipment.
[kg/person]

305 = U17

Source. Bolen, 1980:Ch3

U18 - Mass of OSV guidance equipment. [kg]

185 < U18 < 277.5 nominal = 200

Source. group est.

U19 - OSV mass needs/mission:

assume all needs require low-G launch

[OSV parts, expendables, no fuel or people] [kg]

100 = U19

Source. group est.

U21 - Coef effect of man on the OSV for R&D of the OSV

0 < U21 < 1 nominal = 0.85

Source. group est.

U22 - COST/UNIT OF OSV FUEL [dollars/kg]

\$0.25 < U22 < \$0.40 nominal = \$0.32/kg

Source. Koelle, 1982: 413

Source. Rehder, 1979

U23 - Cumulative ave learning curve slope for OSV

0.95 = U23

Source. Fong, 1981: Ch4

U24 - OSV ISP. [hr]

U24 = 0.14

Source. NASA, 1984d : 42

U25 - LG to OSV mass transfer rate

(LG-OSV and FHG-LG-OSV model)

OSV and SB mass transfer rate (kg/time)
 450 < U25 < 900 kg/hr nominal = 800kg/hr
 Source. Watkins, 1985

U26 - OSV to Satellite mass transfer rate (kg/time)
 225 < U26 < 450 kg/hr nominal = 400kg/hr
 Source. Watkins, 1985

U29 - OSV technical development correction factor
 0.5 < U29 < 1.25
 new system --1.25
 technology exists--0.8 to 1.0
 remake existing system-0.5 to 0.8
 Source. Koelle, 1982: 405 nominal = 1.25

U30 - OSV R&D team experience factor
 0.7 < U30 < 1.3
 1.3=no experience 0.7 previous experience
 Source. Koelle, 1982: 405 nominal = 1.3

U31 - Cost/unit of LG fuel [dollars/kg]
 \$0.25 < U31 < \$0.40 \$/kg nominal = \$0.32/kg
 Source. Koelle, 1982: 413
 Source. Rehder, 1979

U32 - Cumulative average learning curve slope for LG
 0.95 = U32
 Source. Fong, 1981: Ch4

U33 - LG ISP [hr.]
 0.11 = U33
 Source. NASA, 1984d : 54

U34 LG mass structure ratio lower bound
 U34 = 0.1
 Source. Hill and Petterson, 1970: 328

U35 LG to SB mass transfer rate (kg/hr)
 450 < U35 < 900 nominal = 900kg/hr
 Source. Watkins, 1985

U36 LG mass structure ratio upper bound
 U36 = 1.0
 Source. Hill and Petterson, 1970: 328

U37 - LG technical development correction factor
 0.5 < U29 < 1.25
 new system --1.25
 technology exists--0.8 TO 1.0
 remake existing sys-0.5 TO 0.8
 Source. Koelle, 1982: 405 nominal = 1.25

U38 - LG R&D team experience factor
 0.7 < U30 < 1.3
 1.3=no experience. 0.7 previous experience
 Source. Koelle, 1982: 405 nominal = 1.3

U40 - Min LG vehicle required downtime/mission
 (land to launch)
 168 < U40 < 720hrs tav type 168 shuttle type 720
 Source. group est. nominal = 400hrs

U48 - LG launch site minimum time between launches
 24 hrs = U48
 Source. group est

U49 - LG launch site maximum time between launches
 8760hrs = U49
 Source. group est

U58 - Min alt LG must be able to obtain
 185Km = U58
 Source. group est.

U75 - SB life support mass requirements per person time
 [kg/person*hr] (ie air,water,food etc)
 $0.1 < U75 < 1$ nominal = 0.20
 Source. Guy, 1983 note: SB will recirculate

U76 - Life support mass requirements per person time
 [kg/person*hr] (ie air,water,food etc)
 $0.1 < U76 < 1$ nominal = 0.64
 Source. Guy, 1983 note: OSV will not recirculate

U77 - People time in space before rotation [hrs]
 (radiation exposer,physical,labor contract, etc)
 $720 < U77 < 4320$ nominal = 2160
 Source. NASA, 1984b: 400

U78 - Cost/man-time
 $\$1000/\text{man-hr} = U78$
 Source. group est

U79 - Cost/man-year for cost equations
 $U79 = \$125000/\text{man-year 1983 dollars}$
 Source. Koelle, 1982: 402

U90 - Earth Gravitational Constant(μ)
 $= 5.1658716e12[\text{km}^3/\text{hr}^2]$
 Source. Bate, 1970: 429

U91 - Earth Radius 6378.145 km
 Source. Bate, 1970: 429

U92 - PI - 3.141592654
 Source. Bate, 1970: 429

U93 - Earth Gravitational Constant at Earth Surface(g_e)
 $=127137.6$
 Source. Hill and Petterson, 1970: 322

U111 - Cost per unit of earth based energy
 [initial launch energy]
 U111 = \$0.00115/watt-hr
 Source. one-half Dayton Ohio Electrical
 rate of \$2.30/kw-hr]

U112 - Coef for FHG launch site operations cost:
 U112 = \$100000
 Source. group est.

U114 - Coef for cost/unit of FHG apogee fuel burned
 $\$0.25 < U114 < \0.40 nominal $== \$0.32/\text{kg}$
 Source. Koelle, 1982: 413
 Source. Rehder, 1979

U117 - Coef for FHG launch site R&D costs:
 U117 est. $== \$100$
 Source. group est.

U118 - Coef for FHG launch site fabrication/purchase
 U118 est $== \$100$
 Source. group est.

U119 - Coef for FHG power source fabrication/purchase est
 U119 == \$100
 Source. group est.

U120 - Min time between launches from a specific
 FHG launch site
 U120 est. == 24hrs
 Source. group est.

U121 - lsp of FHG fuel used during apogee delta vel
 [use ave chemical]
 U121 =0.09hr
 Source. group est..

U123 - Min mass of payload per FHG launch
 (usable by OMV, satellites)
 U123=100kg
 Source. group est

U129 - % of SB structure mass for fuel calculation/time
 U129 = 0.00000239
 Source. NASA, 1984b: 465

U130 - Coef for SB safty level [length of time] =1month
 U130 = 720
 Source. group estimate

U132 - % of SB structure mass for parts calculation/time
 this represents the total mass of the SB
 replaced every 10.0yrs
 U132 = 0.000011415
 Source. Space station configuration discription

U133 - SB structure mass to mass storage capacity ratio

$0.5 < U133 < 2$ nominal = 1.0

Source. NASA, 1984b: 45

U134 - Cost per unit of mass for SB upkeep (ie. parts)

U134 = \$100.00/kg

Source. group estimate

U135 - Coef for SB R&D manned influence

U130 = \$500000.0

Source. group est

U136 - Coef for SB R&D structure

U130 = \$1000000.0

Source. group est

U137 - Coef for SB production cost

U130 = \$1000000.0

Source. group est

U138 - Coef for SB deployment cost

U130 = 20000/kg

Source. group est

U139 - SB station keeping fuel cost/kg

$\$0.25 < U139 < \0.40 nominal = \$0.32

Source. group est

Intermediate Variables

103 - Launch site designed capability

(designed launch rate) [launches/time]

104 - Launch site operational cost/time (for each site)

105 - LG fuel cost/time

106 - LG R&D cost
 107 - LG production cost
 112 - LG mission time
 113 - LG -time from launch to rendezvous with OSV or SB
 114 - LG docked time
 (time to unload LG payload to OSV or SB)
 115 - LG - return to earth time
 121 - LG delta velocity per mission
 128 - Structural mass ratio of LG
 129 - [LG loaded mass/payload mass]**(1/#LG stages)
 132 - OSV delta vel for resupply (up and back)/OSV mission
 133 - OSV delta vel in waiting orbit (1 sat to 1 sat)
 134 - Semi-major axis of elliptical waiting orbit
 135 - OSV total delta vel/OSV mission
 136 - Velocity at apogee of OSV resupply transfer orbit
 137 - Velocity at perigee of OSV resupply transfer orbit
 140 - OSV req people mass to orbit/unit time
 141 - OSV required mass/time
 142 - Structural mass ratio of OSV
 143 - Mass of req equipment for manned life support
 (nonrobotic needs)
 144 - Total OSV net mass=
 structure+guidance+life support equipment
 145 - OSV fuel cost/time
 146 - OSV manned cost/time
 147 - OSV R&D cost
 148 - OSV production cost

I150 - OSV mission length { time/OSV mission
 =(time to travel from supply orbit to satellites
 orbit and back)/OSV mission
 + [(total time to travel in waiting orbit between
 satellites serviced)
 + (time spent servicing satellites on location)]
 /OSV mission}

I151 - OSV resupply time

I152 - Time to service one satellite

I100 = FHG vehicle vel leaving the ground based launcher

I101 = FHG prelaunch velocity (earth's rotation velocity)

I103 = Actual mass launched from FHG earth launcher

I105 = Min ground based FHG energy/launch
 to provide required delta velocity

I106 = Cost of FHG ground based energy/time

I108 = FHG apogee delta velocity required

I109 = Required FHG parking orbit velocity
 (so FHG will stay in rendezvous orbit)

I110 = FHG velocity prior to apogee burn

I111 = Parking [rendezvous] altitude from earth's center

I113 = Mass of fuel required at the FHG apogee burn

I115 = FHG launch site operations cost/time

I116 = FHG vehicle costs/time

I118 = FHG vehicle R&D costs

I119 = FHG launch site R&D
 (launcher and power source or plant)

I120 = FHG launch site purchase or fabrication costs

1124 = SB fuel required/time (all SB's in operation)
1125 = SB parts mass required/time
 (parts, nonfuel, or nonlife support)
1126 = SB life support mass required/time
1127 = SB people mass/time
1130 = SB manned cost/time
1131 = SB replentishment costs/time (fuel+parts+etc only)
1134 = SB R&D costs
1135 = SB production-fabrication cost
1136 = SB deployment costs

Appendix F

Intermediate Variable Equations

The intermediate variables used in the model are for algebraic simplification only. When possible a name is given to the variable as an aid in understanding the equation the variable represents. When a literature source was identified for an equation the source is provided. In other cases when no literature could be found the group derived an equation; therefore, as future work and research continues the equations should be updated.

I03 = LG Launch Site Designed Capability

I03 = 1.5/X335

Source. Group Derived [assumed design rate = 1.5 req rate]

I04 = LG Launch Site Operational Cost

I04 = 40*(U79/8760)*(I03**0.34)/[(1/X335)**0.55]

*(X320)/X335

= [40*(man year cost/hrs in yr)

*(launch site designed capacity**0.34)

*(1/(launch site launch rate**0.55))]

*(# launch sites)/(time between launches at a site)

Source.(Koelle, 1982:417) cost in 1983 dollars

I05 = LG Fuel Cost/Time

I05 = X326*U31*X301

= fuel/mission *(cost/unit of fuel) *(#LG missions)

Source. Group Derived

I06 = LG R&D Cost

$$\begin{aligned} I06 &= 6500 * U79 * [X325 ** 0.21] * U37 * U38 * [\exp(2 * X345)] \\ &= 6500 * U79 * (\text{empty mass weight} ** 0.21) * \text{tech factor} \\ &\quad * \text{experience factor} \\ &\quad * [\text{cost multiplier for increased reliability}] \end{aligned}$$

Source. (Koelle, 1982: 407) cost in 1983 dollars

I07 = LG Production Cost

$$\begin{aligned} I07 &= X300 * 12 * [X325 ** 0.56] * [X300 ** \{\ln(U32) / \ln(2.0)\}] \\ &\quad * U79 * [\exp(2 * X345)] \\ &= (\#LGs) * ((16.5 + 7.5) / 2) * ((X325) ** 0.56) \\ &\quad * ((\#LG * (\ln U32 / \ln 2)) * (\text{cost per man-year})) \\ &\quad * [\text{cost multiplier for increased reliability}] \end{aligned}$$

Source. Koelle, 1982: 409 cost in 1983 dollars

Source. Rand, 1971: Ch 5

I12 = LG Mission Time

$$I12 = I13 + I14 + 15$$

Source. Group Derived

I13 = LG Time from Launch to Rendezvous with OSV

$$I13 = \{\text{SQRT}[U90 / (X360 + U91)] + 5486 + 1152 + 1097\} / (2 * U93)$$

Source. Dept of AF, 1965: 2-46

I14 = LG Docked Time (time to offload the LG

payload to OSV or SB)

$$I14 = X310 / U25 \text{ (for LG+OSV and FHG+LG+OSV model)}$$

$$I14 = X310 / U35 \text{ (for SB+FHG+LG+OSV model)}$$

Source. Group Derived

I15 = LG Return Time to Earth

I15 = 3hrs

Source. Group Derived

I21 = LG Delta Velocity per Mission

I21 = $\text{SQRT}[U90/(X360+U91)]+5486+1152+1097+0$

Source. Dept of AF, 1965: 2-46

I28 = LG Structural Mass Coefficient

I28 = $X325/(X325+X326)$

Source. Hill and Peterson, 1970: 328

I29 =

I29 = $X326+X310)/X310]**(1/X370)$

= $[LG \text{ loaded mass/payload mass }]**(1/\#LG \text{ stages})$

Source. Hill and Peterson, 1970: 328

I32 = OSV Delta Velocity (orbit transition from resupply
point to satellite orbit back to resupply orbit)

I32 = $2*ABS(\text{SQRT}(U90/(U1+U91))-\text{SQRT}(U90/(X360+U91)))$

Source. Appendix G, EQ(G.11)

I33 = OSV Delta Velocity in waiting orbit (1 sat to 1 sat)

I33 = $2*ABS((\text{SQRT}(U90/(U1+U91))-\text{SQRT}(2*U90/(U1+U91)-U90/I34)))$

Source. Appendix G, EQ(G.24)

I34 = Semi-major Axis of Elliptical Waiting Orbit

I34 = $(U1+U91)*((U2*X565-1)/(U2*X565))**(2./3.)$

Source. Appendix G, EQ(G.19)

I35 = OSV Total Delta Velocity/OSV mission

I35 = I33*(X560-1)+I32

Source. Appendix G, EQ(G.25)

I36 = Velocity at Apogee of OSV Resupply Transfer Orbit

I36 = SQRT(2*ABS(U90/(U1+U91)-U90/(X360+U1+2*U91)))

Source. Appendix G, EQ(G.8)

I37 = Velocity at Perigee of OSV Resupply Transfer Orbit

I37 = SQRT(2*ABS(U90/(X360+U91)-U90/(X360+U1+2*U91)))

Source. Appendix G, EQ(G.6)

I40 = OSV - People Mass to Orbit/Unit Time

I40 = X500*X555*100/U77

Source. Group Derived

I41 = OSV Required Mass/Time

I41 = [(X526+U19)*X501]+(X555*U76*X500)+I40

= [(fuel+parts)*#missions]

+((life support)+(people mass))

Source. Group Derived

I42 = OSV Structural Mass Ratio (0.1<I42<1)

I42=X525/(X526+X525)

Source. Hill and Peterson, 1970: 328

I43 = Mass of Life Support Equipment for Manned OSV

I43 = U16 + U17*X555

Source. Bolen, 1980: Ch 3

I44 = OSV Single Vehicle Mass (dry weight)

I44 = X525+ U18+ I43

= structure+guidance+life support equipment

Source. Group Derived

I45 = OSV Fuel Cost/Time

I45 = X526*U22*X501

= fuel burned/mission*(cost/unit of fuel)

*(#OSV missions)

Source. Group Derived

I46 = OSV Manned Cost/Time

I46 = X500*X555*U78

= (#OSV's)*(OSV crew size)*(cost/mantime)

Source. Group Derived

I47 = OSV R&D Cost

I47 = 6500*U79*[I44**0.21]*U29*U30*((U21)**X555]

*[exp(2*X545)]

= 6500*cost per man-year

*(total empty mass weight**0.21)

* tech factor* experience factor

* [multiplier for addition of man in the OSV]

* [cost multiplier for increased reliability]

Source. (Koelle,1982:407)

in 1983 dollars

I48 = OSV Production Cost

I48 = X500*[16.5]*[I44**0.56]*[X500**((ln(U23)/ln(2.0)))]

U79[exp(2*X545)]

$$= (\#OSV) * ((16.5) * (144 ** 0.56))$$

* (cumulative ave learncurve) * cost per man year

* [cost multiplier for more reliability]

Source. Koelle, 1982: 407 in 1983 dollars

Source. Rand, 1971: Ch 5

I50 = OSV Mission Length

= (time to travel from supply orbit to satellite
orbit and back) / OSV mission

+ [(total time to travel in waiting orbit
between satellites serviced)

+ (total time spent servicing satellites
on location)] / OSV mission

$$I50 = 2 * U92 * (((2 * U91 + U1 + X360) / 2.) ** 1.5) / \text{SQRT}(U90)) \\ + (X560 - 1) * (2 * U92 * ((U1 + U91) ** 1.5) * (U2 * X565 - 1.) \\ / (U2 * \text{SQRT}(U90))) + X560 * I52$$

Source. Appendix G, EQ(G.35)

I51 = OSV Resupply Time

$$I51 = X510 / U25$$

Source. Group Derived

I52 = Time to Service One Satellite

$$I52 = X561 / U26$$

= mass delivered

/ OSV to Satellite mass transfer rate

Source. Group Derived

I100 = Post Launch FHG Vehicle Velocity

(ie. velocity leaving the ground based launcher)

$$I100 = \text{SQRT}(2 * \text{ABS}[U90/U91 - U90/(2 * U91 + X360)])$$

(note: this represents the speed the mass (vehicle) would be traveling to be at the perogee of an elliptical orbit.)

Source. Appendix G, EQ(G.6)

I101 = FHG launcher velocity.

(speed launcher is traveling due the earths rotation. note our launcher is considered to be physically located on the equator.)

$$I101 = U91 * 2 * U92 / 24$$

$$= U91 * U92 / 12$$

$$= (\text{radius}) * (\text{angular vel})$$

$$= (\text{earth radius at equator}) * 2$$

$$* \pi / (\text{rotation period earth})$$

Source. Appendix G, EQ(G.5)

I103 = Total FHG Vehicle launch mass

$$= X110 + 10 * I113 / 9$$

$$= (\text{mass payload}) + (10/9) * (\text{mass fuel})$$

[note: mass structure = mass fuel/9]

Source. Group Derived

1105 = Ground Based FHG Energy for Launch

(note: for simplicity we will consider only the
change in kinetic energy of the mass launched
from earth)

$$= 4.629 * 0.5 * 1103 * (1100 - 1101) ** 2$$

$$= 2.3145 * 1103 * (1100 - 1101) ** 2$$

$$= [one-half * mass * delta vel ** 2]$$

*conversion factor to get 1105 in watt-hrs (4.629)

Source. Group Derived

1106 = Cost of FHG Ground Based Energy/Time

(note: consider an electrical source)

(computed based on twice the minimum
required energy from 1105)

$$= X100 * (2 * 1105) * U111 = cost/time$$

Source. Group Derived

1108 = FHG Apogee Delta Velocity Required

$$= 1109 - 1110$$

Source. Appendix G, EQ(G.10)

1109 = Required FHG Parking Orbit Velocity

(so FHG will stay in rendezvous orbit)

$$= \text{SQRT}\{U90/1111\}$$

Source. Appendix G, EQ(G.9)

1110 = FHG Velocity Prior to Apogee Burn

(ie velocity from initial launch)

$$= \text{SQRT}\{2 * \text{ABS}[U90/I111 - U90/(U91 + I111)]\}$$

Source. Appendix G, EQ(G.8)

I111 = Altitude of Parking Orbit

(rendevous altitude from the center of the earth)

$$= U91 + X360$$

Source. Group Derived

I113 = Mass of Fuel Required at FHG Apogee Burn

$$= X110 * \{ \exp(I108/(U121 * U93)) \}$$

$$/ [10/9 - \exp(I108/(U121 * U93))/9] \}$$

(note: this equation takes into consideration the mass of the unloaded vehicle by assuming the mass structure ratio will be 0.1. This assumption is implemented in the 10/9 and the exp ()/9.

Source. Group Derived

I115 = FHG Launch Site Operations Cost/Time

$$= X120 * U112 * (X135^{*-2})$$

Source. Group Derived

I116 = FH Vehicle Costs/Time

(note: assume FHG Vehicle is expendable)

$$= X100 * \{ (1.057 * 0.0624534 * I103 * I108) \}$$

$$+ (46330 * [(0.4536 * I113/9) * 0.77]) + (U114 * I113) \}$$

$$= \# \text{FHG launches/time}$$

$$* \{ \text{engine cost} + \text{platform cost} + \text{fuel cost} \}$$

$$= \# \text{FHG launches/time} * \{ (\text{cost coef}) \}$$

$$* (\text{conversion kg-km/hr to lbf-sec})$$

$$\begin{aligned}
 & * (\text{mass to delta vel}) * (\text{delt vel req}) \\
 & + [\text{cost coef}] * (\text{kg to lfm conv} * \text{platform mass}) ** .77 \\
 & + [\text{FHG fuel cost}]
 \end{aligned}$$

Source. Fong, 1981: Ch 5

$$\begin{aligned}
 \text{I118} &= \text{FHG Vehicle R\&D Costs} \\
 &= \{[3.235 * 0.0624534 * \text{I103} * \text{I108}] \\
 &\quad + [7414460 + 22600 * 0.4536 * \text{I113}/9]\} \\
 &= \{\text{engine} + \text{platform R\&D}\}
 \end{aligned}$$

Source. Fong, 1981: Ch 5

$$\begin{aligned}
 \text{I119} &= \text{Launch Site R\&D} \\
 &= \text{U117} * \text{X360} * (\text{I103} ** 0.5)
 \end{aligned}$$

Source. Group Derived

$$\begin{aligned}
 \text{I120} &= \text{Launch Site Purchase or Fabrication} \\
 &= \text{X120} * \{[\text{U118} * (\text{I103} ** 0.5)] \\
 &\quad + [\text{U119} * (2 * \text{I105}) ** 0.5]\}
 \end{aligned}$$

Source. Group Derived

$$\begin{aligned}
 \text{I124} &= \text{SB Fuel Mass Required per Time} \\
 &= \text{X400} * \text{X425} * \text{U129} \quad (\text{note: could be high-G launched})
 \end{aligned}$$

Source. Group Derived.

$$\begin{aligned}
 \text{I125} &= \text{SB Mass Requirements/Time (require low-g launch)} \\
 &= \text{X400} * \text{X425} * \text{U132} \\
 &= [\text{\#SB's}] * [\text{SB structure mass}] \\
 &\quad * [\% \text{ SB structure mass needing replacement per time}]
 \end{aligned}$$

Source. Group Derived.

I126 = SB Life Support Mass Required/Time

$$= X400 * U75 * X415$$

$$= [\#SB's] * [\#people/SB] * [mass/person]$$

(note: assume SB's recirculate life support mass
and all life mass is low-G launched)

Source. Group Derived.

I127 = SB People Mass/Time

$$= X400 * X415 * 100 / U77$$

Source. Group Derived

I130 = SB Manned Cost/Time

$$= X400 * X415 * U78$$

Source. Group Derived

I131 = SB Replenishment Costs/Time (for fuel+parts)

$$= I125 * U134 + I124 * U139$$

$$= [replenishment\ mass/time] * [cost\ of\ mass]$$

Source. Group Derived

I134 = SB R&D Costs

$$= U136 * \{X425 ** 0.5\} + U135 * \{X415 ** 0.75\}$$

Source. Group Derived

I135 = SB Production-Fabrication Cost

$$= X400 * U137 * [X425 ** 0.5]$$

Source. Group Derived

I136 = SB Deployment Costs

$$= X400 * U138 * X425$$

$$= [\#SB's] * [launch\ cost/kg] * [SB\ structure\ mass]$$

Source. Group Derived

Appendix G

Derivation of OSV Orbital Mechanics Equations

The size of an OSV is determined primarily by two factors: the cargo mass it must deliver to the satellite(s) and the fuel mass the OSV must deliver and consume to accomplish the mission. Here we present the development of the equations used to model the dynamics of the OSV. Since the SSS models require OSV quantities on a per mission basis, the following development is for a single OSV mission. First we will begin by defining an OSV mission and present the assumptions made to simplify the form of the equations. Then the equations will be developed for the OSV fuel consumption, elliptical orbits, transfer orbit, waiting orbit, and mission length/time of flight (TOF). Finally, a brief summary of the primary equations is given; included is a cross-reference table to the intermediate variable equations (Appendix F) and associated model equations (Appendix D).

G.1 OSV Mission Definition

For development of the Satellite Servicing System (SSS) model, the mission of an OSV is defined as: (1) loading on supplies while in some circular parking orbit, (2) transferring from the parking orbit to the servicing (satellite) orbit, (3) delivering an average amount of mass to each satellite, and (4) returning empty to the parking orbit.

6.2 Simplifying Assumptions

To develop the analytical equations that describe the OSV mission, the following simplifications and assumptions were made:

1. 100 percent of an OSV payload capacity is distributed per mission.
2. The parking orbit is the same at the beginning and end of a mission, however, it is free to be chosen within the model.
3. The satellite and parking orbits are circular.
4. There are no perturbations of orbits (transfer, parking, or servicing).
5. The effect on satellite orbits from oblateness of the earth is neglected.
6. The drag of the atmosphere and solar wind on OSV is neglected.
7. Station keeping maneuvers by OSV are not considered.
8. Rendezvous maneuvers are neglected; the rendezvous is assumed to occur immediately at intercept of transfer orbit and target orbit.
9. Satellites are equally spaced within the orbit plane.
10. No priority servicing of satellites within constellation. Therefore, the OSV will travel from satellite to satellite in a consecutive order.
11. The maneuver used to travel between satellites within the same constellation will be the same between any two satellites.
12. The requirements of satellites in a constellation will be averaged to allow the same amount of mass delivered to each. Or looked at another way, satellites within a constellation are identical and therefore require the same amount of mass when serviced.
13. In order to keep the mission time length short, the OSV propulsion system is assumed to be chemical.

G.3 OSV Fuel Consumption

Since the type of propulsion system is a chemical (solid or liquid), the fuel mass used can be calculated from the Rocket equation (Hill, 1970):

$$\Delta V_{osv} = g_e * I_{sp} * \ln(M_i / M_o) \quad (G.1)$$

where

g_e = gravitational acceleration at surface of the earth (9.807 m/sec²)

I_{sp} = specific impulse of fuel(sec)

M_i = initial total vehicle mass(kg)

M_o = final total vehicle mass(kg)

ΔV_{osv} = magnitude of total change in velocity of OSV to travel between two points(m/sec)

Since the fuel mass needed for the model is that per OSV mission, then the final and initial masses of the OSV are defined as:

$$M_o = M_s + M_p \quad (G.2)$$

$$M_i = M_s + M_p + M_f$$

$$M_i = M_o + M_f \quad (G.3)$$

where

M_s = OSV structural(dry) mass (kg)

M_p = total payload(supply) mass delivered to satellites per OSV mission (kg)

M_f = fuel mass (kg)

Substituting Eq (G.3) into Eq (G.1) and rearranging, the fuel mass used per OSV mission in terms of the OSV final (empty) total mass and the payload mass is:

$$M_f = M_o \{ \exp[\Delta V_{osv} / (g_e \cdot I_{sp})] - 1 \} \quad (G.4)$$

The fuel mass per mission calculated by using Eq (G.4) is conservatively high. The equation is derived assuming that the payload mass is removed from the OSV all at once at the end of the mission, rather than dropping mass off at each satellite. If this staging process were to be modelled, an iterative application of Eq (G.4) would be used with new values for M_i and M_o determined at each start/stop point of the OSV mission. Then the fuel mass used for each leg of the mission is totalled to give the total fuel mass. A FORTRAN program called VORBCS.F is developed to calculate the fuel mass taking into account staging (see Appendix H). Table G.1 shows a comparison of the fuel mass calculations from this program with those using the approximation of Eq (G.4). The values in the first four columns describe the mission and are used to calculate ΔV_{osv} . These equations will be described later. However, notice that the estimated fuel mass is indeed conservatively high and the difference between the fuel mass values does increase with increasing average mass per satellite and number of satellites serviced. This is expected because the effect of staging saves more fuel mass for a given ΔV_{osv} as the number of off-loading events increases. In Chapter V of this study

Table G.1
Fuel Consumption Comparisons

Alt (km)	# of Sats		Mass (kg)	Δ Vosv (km/hr)	Fuel Mass/Mission (kg)		% difference
	Total N	Serviced Y			Eq(G.4) Mf(1)+	VORBCS.F Mf(2)+	
800	144	5	50	2955.659	460.055	433.325	6.169
800	144	5	100	2955.659	511.172	457.713	11.680
800	144	5	350	2955.659	766.758	579.651	32.279
800	144	5	500	2955.659	920.110	652.815	40.945
800	144	5	250	2955.660	664.524	530.876	25.175
800	144	36	250	6832.492	5910.829	3271.129	80.697
800	144	72	250	11334.620	20819.892	10142.888	105.266
800	144	108	250	15836.749	49579.549	22214.354	123.187
800	144	144	250	20338.888	98698.918	41188.989	139.625

+ Number of waiting orbits (n) set = 1
 Altitude of parking orbit set = 185.2 km
 OSV structural mass (Ms) set = 2000 kg
 lsp set = 450 sec

is addressed the effect of using Eq (G.4) as an approximation by examining the sensitivity of the final results to changes in total fuel mass consumed.

In order to use Eq (G.4), the ΔV_{OSV} must be calculated. This is dependant on how an OSV moves in space. We treated the motion of the OSV relative to the center of the earth as a classical two body problem in orbital mechanics. The mission dynamics of the OSV were broken up into two distinct phases: (1) transfer to and from the service orbit from and to the parking orbit; and (2) transfer between satellites of the same orbit, i.e. intersatellite maneuvers. But before we discuss the ΔV calculations for the above two phases we must define some elements of the two-body problem; namely the elliptical orbit elements.

G.4 Elements of Elliptical Orbits.

For a small body (relative to earth's mass) in free flight orbit near the earth, the two body problem predicts that the trajectory of the body will be that of an ellipse with the center of the earth at one focus (Bate, 1971:30). Figure G.1 shows the elliptical orbital elements.

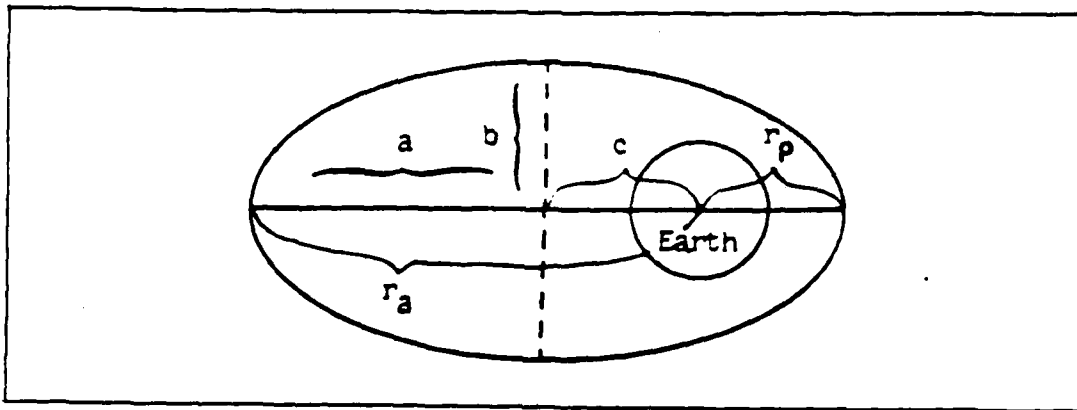


Figure G.1 Elliptical Orbit Elements (Dept. of AF. 1985:2-15)

where

a = semi major axis

b = semi minor axis

c = distance from origin to either focus

r_a = radius of apogee

r_p = radius of perigee

Notice that circular orbits are a special case of elliptical orbits when $r_a = r_p = r$.

Since the OSV is assumed to have chemical propulsion, the forces used to change its velocity will be considered impulsive. Therefore, after an instantaneous change in velocity, the OSV will be in free flight about the earth and consequently in an elliptical orbit.

G.5 Transfer Orbit

There are many ways an OSV could change from one circular orbit to another via an elliptical transfer orbit. The

choice depends on a trade-off between ΔV and transfer time. The quicker one wishes to change between orbits the more ΔV (and therefore fuel) the vehicle must expend. As a first modeling attempt for this study, minimizing fuel consumption was felt more important than minimizing transfer time.

Therefore, for the case of transferring between two coplanar circular orbits a Hohmann transfer was modeled. This maneuver is shown in figure G.2 and requires the minimum total ΔV (Bate, 1971:163).

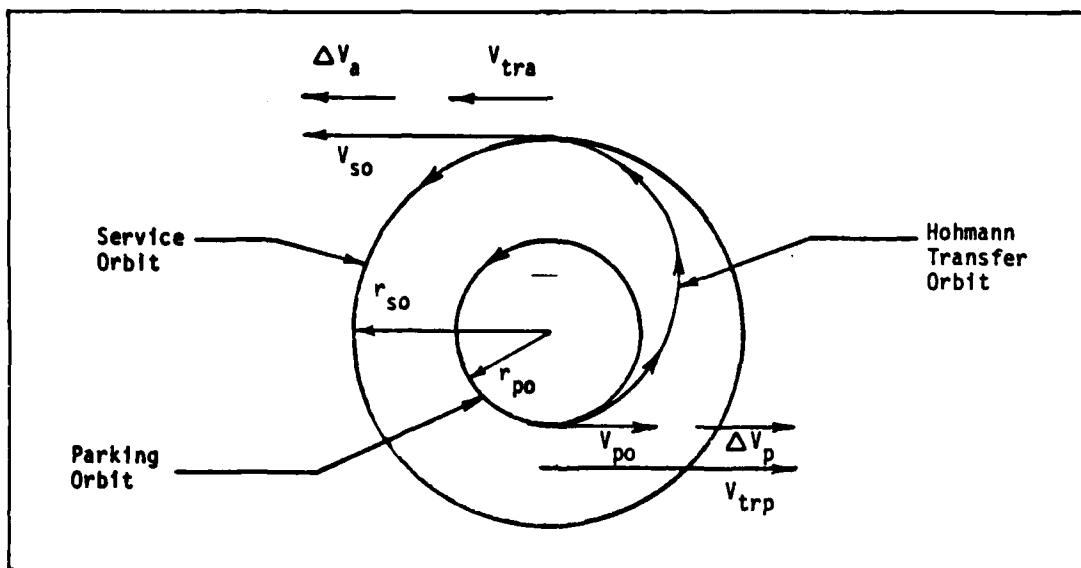


Figure G.2 Hohmann Transfer between Coplanar Circular Orbits

The Hohmann transfer trajectory is an ellipse such that the perigee is tangent to the inner circular orbit ($r_p = r_{po}$) and the apogee is tangent to the outer circular orbit ($r_a = r_{so}$). Two impulses (or burns) are required to complete the maneuver.

Assume the transfer is from the parking (inner) orbit to the servicing (outer) orbit. Figure G.2 depicts the orbital velocities relative to the center of the earth. In order for the OSV to initiate travel along the transfer ellipse at perigee it must increase its speed by ΔV_p . The OSV velocity in the circular parking orbit is :

$$V_{po} = (\mu/r_{po})^{1/2} \quad (G.5)$$

where

μ = Gravitational parameter, $3.986032 \times 10^5 \text{ km}^3/\text{sec}^2$

r_{po} = Radius of parking orbit

and the OSV velocity in the elliptical transfer orbit at perigee would be:

$$V_{trp} = \{2[\mu/r_{po} - \mu/(\bar{r}_{po} + r_{so})]\}^{1/2} \quad (G.6)$$

where

r_{so} = Radius of the service orbit.

Therefore, for the first impulse to get the OSV into the transfer orbit:

$$\Delta V_p = |V_{trp} - V_{po}| \quad (G.7)$$

The second impulse is used to recircularize the OSV orbit into the service orbit when the OSV is at apogee. The velocity of the OSV at apogee is given by:

$$V_{tra} = [2\mu/r_{so} - \mu/(\bar{r}_{po} + r_{so})]^{1/2} \quad (G.8)$$

and the velocity of the circular service orbit is :

$$v_{so} = (\mu/r_{so})^{1/2} \quad (G.9)$$

therefore

$$\Delta v_a = |v_{so} - v_{tra}| \quad (G.10)$$

So the total change in velocity of the OSV to transfer one way between two orbits is $\Delta v_p + \Delta v_a$. Therefore, to go up to the service orbit, recircularize, go back to the parking orbit, and recircularize, the total change in velocity is:

$$\begin{aligned} \Delta v_{resupply} &= 2|\Delta v_a + \Delta v_p| \\ &= 2|v_{so} - v_{tra} + v_{trp} - v_{po}| \end{aligned} \quad (G.11)$$

The above equation is good only for orbits in the same plane, i.e. zero inclination difference between them. For orbits at different inclinations the choice of transfer orbit is not as simple.

The three most common transfers between inclined circular orbits are the bi-elliptic transfer (Figure G.3), the "modified" Hohmann and the Hohmann transfer with plane change (Figure G.4).

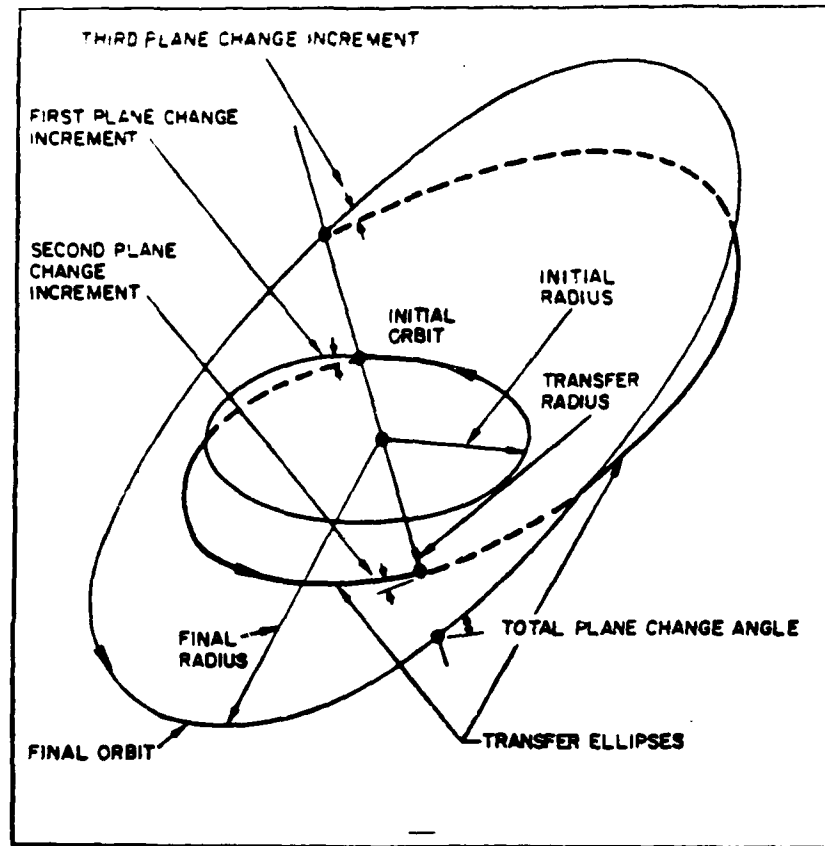


Figure G.3 Bi-elliptic transfer orbit (Dept of AF, 1965)

For orbit transfer without rendezvous the Hohmann transfer with plane change uses the minimum ΔV . But since the OSV must begin and end at the line of nodes, a satellite must be at the precise angle so that both vehicles arrive at the other node simultaneously. The bi-elliptic transfer and "modified" Hohmann don't have this problem but do use more ΔV . The phasing time increases as the ratio of the final to initial orbit radius approaches unity (Baker, 1965:12). However, it is assumed that the ratio of parking orbit radius to service orbit radius will not be near unity for

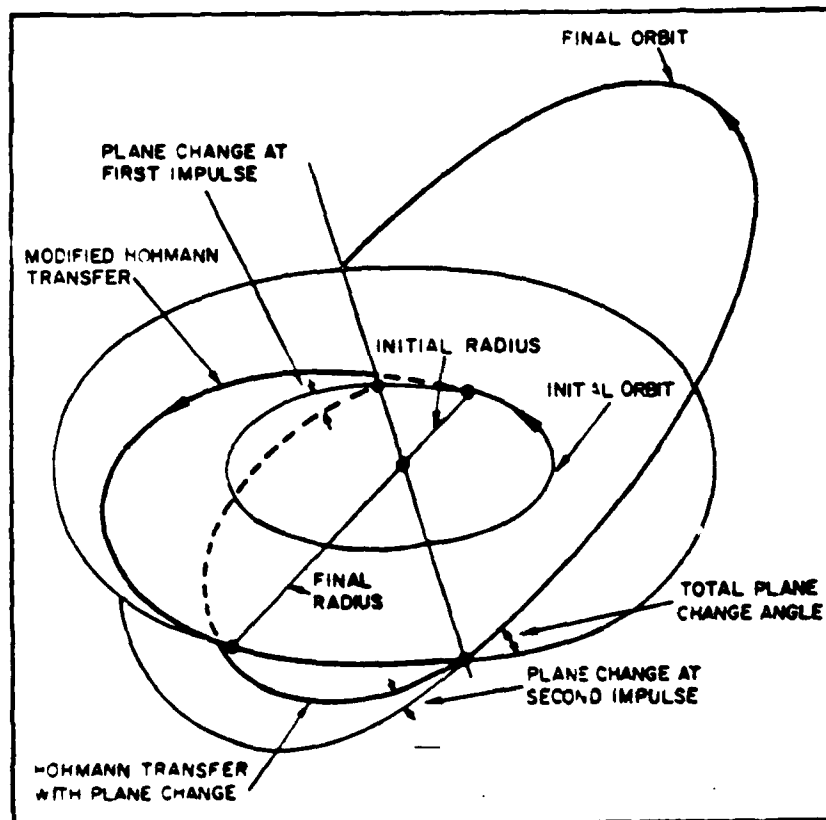


Figure G.4 "Modified" Hohmann and Hohmann transfer with plane change. (Dept of AF, 1965)

our SSS model. In addition, the first satellite to be serviced in an orbit is arbitrary, so the phasing time is again reduced. Since it was felt that minimizing fuel consumption (or ΔV) was very important, we chose the Hohmann transfer with plane change to model transfers between inclined circular orbits. If OSV time of flight becomes more critical, then a choice between the other two techniques is recommended. See (Baker, 1965) for a detailed discussion in this area.

The Hohmann transfer with plane change is a two burn maneuver optimally splitting the inclination change at each burn to achieve minimum ΔV . The first burn usually includes an inclination change of less than 5 degrees with the rest occurring at the second burn.

To calculate the required ΔV at each burn refer to figure G.5. This shows the velocity vectors at a point of intersection between two orbits, and the ΔV required to change from orbit 1 to orbit 2. Once the angle α is known, the ΔV is calculated using the law of cosines as shown.

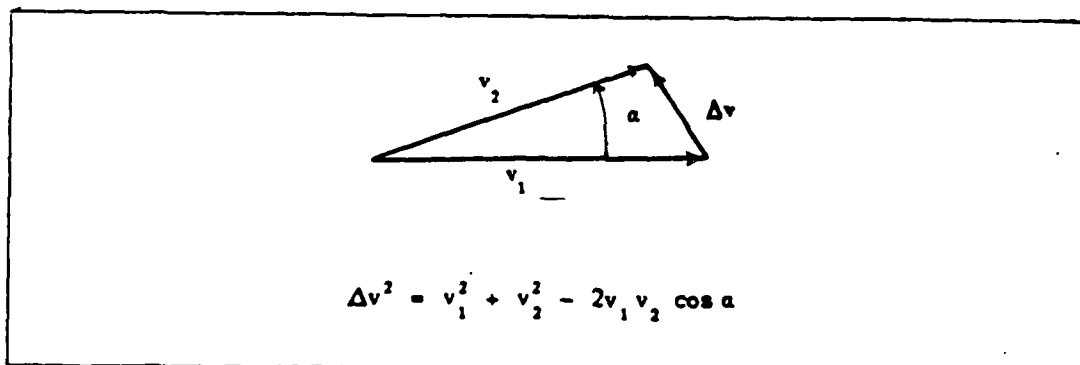


Figure G.5 Velocity Vector Diagram (Dept of AF. 1985:2-40)

Each burn of the Hohmann transfer with plane change combines a plane change maneuver and an altitude change maneuver into one. Figure G.6 shows the vector diagram of such a maneuver at the intersection of a transfer ellipse (begun at 100 nm) and a final circular orbit at 1500nm with an inclination difference of 10 degrees.

At the line of nodes, α in Figure G.6 is the inclination difference between the two orbits. Let β be the amount

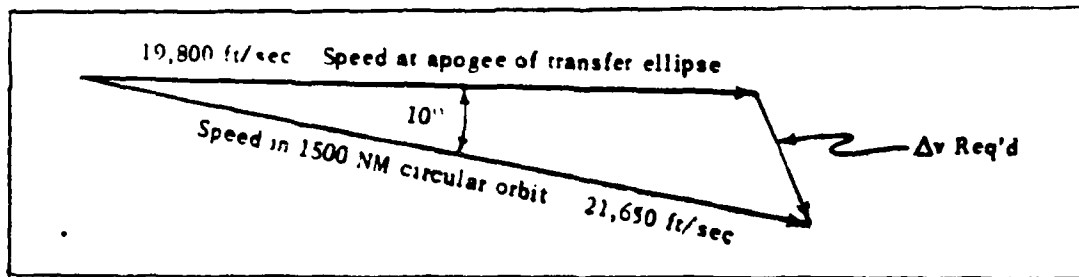


Figure G.6 Combined Maneuver Vector Diagram
(Dept of AF, 1985:2-41)

of plane change (less than 5 degrees) at the first burn and θ be the total plane change. Then the change in velocity at the first burn is :

$$\Delta v_1 = [v_{po}^2 + v_{trp}^2 - 2v_{po}v_{trp} \cos(\beta)]^{1/2} \quad (G.12)$$

and at the second burn

$$\Delta v_2 = [v_{tra}^2 + v_{so}^2 - 2v_{tra}v_{so} \cos(\theta - \beta)]^{1/2} \quad (G.13)$$

where v_{po} , v_{trp} , v_{tra} , and v_{so} are defined by Eqs (G.5), (G.6), (G.8), and (G.9) respectively. Therefore, the total change in velocity between inclined orbits for up, recircularize, back and recircularize is:

$$\Delta v_{resupply} = 2|\Delta v_1 + \Delta v_2| \quad (G.14)$$

Notice when θ is equal to zero this equation reduces to the co-planar case Eq (G.11). One of these two equations ((G.11) or (G.14)) is used to model the Δv used for phase 1 maneuvers. Now let us turn our attention to modeling phase 2 maneuvers: intersatellite transfers.

G.6 Waiting Orbits

Once the USV is in the circular servicing orbit and finishes servicing the satellite it rendezvoused with, it must move on to the next satellite. This is accomplished through an epoch change maneuver (Dept of AF, 1965). This maneuver uses the same principle as the coplanar Hohmann transfer. An impulse is used to place the USV into an elliptical orbit. Since the period of the ellipse will be different than the period of the circular satellite orbit, the OSV and satellites will move with respect to one another. Figure G.7 shows the case for an ellipse outside the circular orbit.

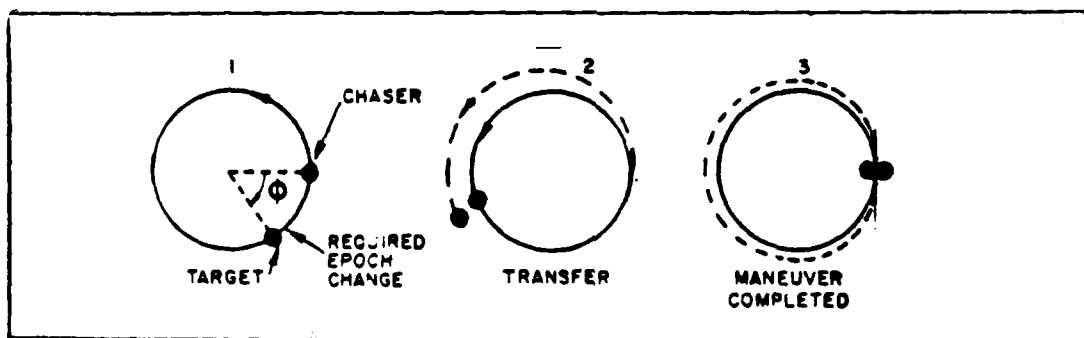


Figure G.7 Epoch Change Using Outside Transfer Orbit
(Dept of AF, 1965)

The size of the ellipse is chosen so a satellite and the OSV rendezvous after an integer number of elliptical orbits by the OSV. Theoretically, very little impulse can be used on the first burn and it would take an infinite number of revolutions of the OSV before it could dock with a satellite. Conversely, a large initial impulse could be used to

achieve rendezvous after one elliptical orbit. Here again, the trade-off between ΔV and time must be made. However, the minimum ΔV requires a stiff penalty in infinite time. Therefore a decision to model the trade-off was made.

An epoch change can occur using an ellipse inside or outside the service orbit. For every exterior ellipse there is an interior ellipse that requires the same total ΔV . The choice of which ellipse is determined by whether the OSV must "catch up to" or "wait for" the next satellite. Since there is no predetermined order in our model we chose to investigate using an interior ellipse because it has the smaller orbital period. This choice requires that precautions be taken to avoid choosing an ellipse which could result in the OSV impacting the earth. Figure G.8 shows the geometries used for the equation derivations.

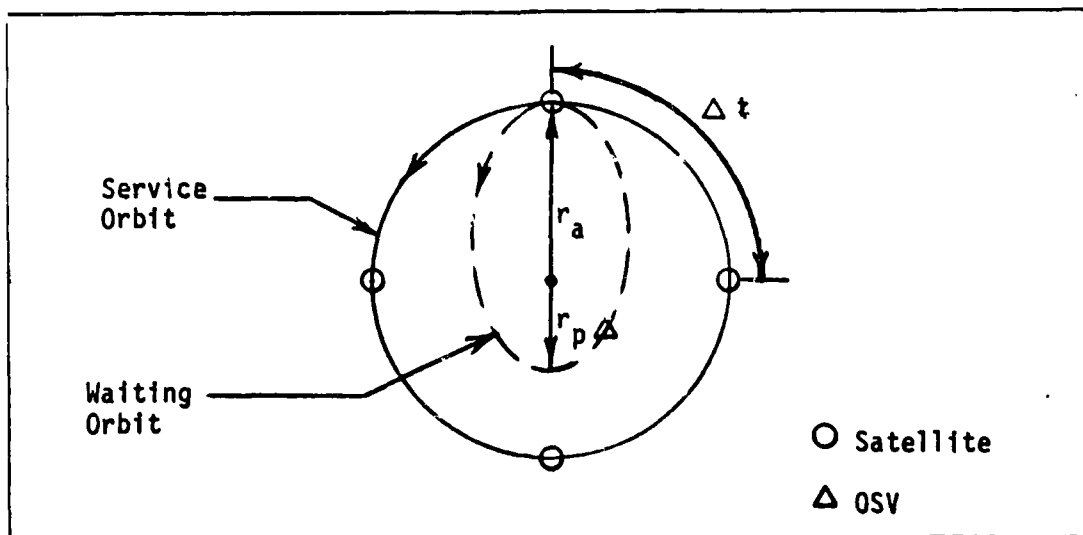


Figure G.8 Waiting Orbit Geometry

In order to assure rendezvous after an integer number of OSV waiting orbits the following must hold:

$$\Delta t = n(T_{so} - T_{wo}) \quad (G.15)$$

where

Δt = Characteristic time between satellites
(see Figure G.7)

n = Integer number of elliptical waiting orbits

T_{so} = Period of one service orbit

T_{wo} = Period of one waiting orbit

For equally spaced satellites $\Delta t = T_{so}/N$, where N is the number of satellites in the orbit. Using this in Eq (G.14) and rearranging, we get:

$$T_{wo} = T_{so}[(Nn-1)/Nn] \quad (G.16)$$

Which clearly shows the dependance of the size of the waiting orbit to n ; the number of waiting orbits before rendezvous. The period of each orbit is related to the orbit geometry as follows:

$$T_{so} = 2\pi r_{so}^{3/2}/\sqrt{\mu} \quad (G.17)$$

$$T_{wo} = 2\pi a^{3/2}/\sqrt{\mu} \quad (G.18)$$

where

a = Semi-major axis of waiting orbit

Using these two expressions in Eq (G.16) and solving for a we get:

$$a = r_{so}[(Nn-1)/Nn]^{2/3} \quad (G.19)$$

since

$$\begin{aligned} 2a &= r_a + r_p = r_{so} + r_p \\ r_p &= 2a - r_{so} \end{aligned} \quad (G.20)$$

This can be used to prevent letting the USV crash into the earth by placing bounds on the adius at perigee:

$$r_{earth} + 185 \text{ km} < r_p < r_{so} \quad (G.21)$$

To calculate the ΔV to insert the USV into an elliptical waiting orbit and recircularize it after n ellipses refer to Figure G.9

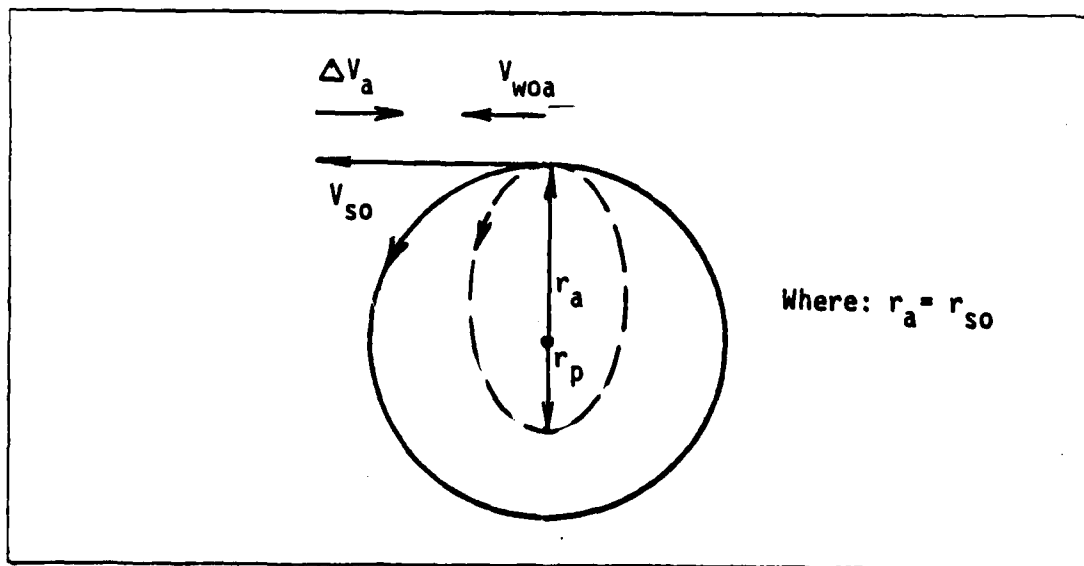


Figure G.9 Waiting Orbit Velocities

The velocity of the service orbit is repeated for convenience:

$$v_{so} = (\mu/r_{so})^{1/2} \quad (G.9)$$

The velocity of the waiting orbit at apogee is :

$$v_{woa} = \{2[\mu/r_{so} - \mu/(r_{so}+r_p)]\}^{1/2} \quad (G.22)$$

Therefore, the change in velocity required to place the OSV into a waiting orbit is:

$$\Delta v_a = v_{so} - v_{woa} \quad (G.23)$$

Noting that $r_{so}+r_p = 2a$ the total change in velocity to get into a waiting orbit and get out of it is:

$$\Delta v_{wo} = 2[(\mu/r_{so})^{1/2} - \{2[(\mu/r_{so}) - (\mu/2a)]\}^{1/2}] \quad (G.24)$$

So Δv_{wo} represents the total Δv to travel from one satellite to the next. Therefore, if Y represents the number of satellites serviced in the orbit, and $Y < N$, then the total Δv per mission is:

$$\Delta v_{osv} = \Delta v_{resupply} + (Y-1)\Delta v_{wo} \quad (G.25)$$

where $\Delta v_{resupply}$ is given by either Eq (G.11) or Eq (G.14) and Δv_{wo} is given by Eq (G.24). These equations represent an OSV mission profile of using a Hohmann transfer with plane change between orbits and an epoch change using an inner ellipse for intraorbit changes to service Y satellites. Using this profile an average mission duration can be estimated for the SSS model.

G.7 Mission Length/Time of Flight (TOF)

The time length per OSV mission can be conveniently broken up into three parts: (1) phasing/aligning time, (2) transfer time to and from service orbit and (3) time spent travelling between and servicing Y satellites in the constellation.

The phasing time is the time after receiving supplies that the OSV must loiter in the parking orbit until it can initiate orbit transfer. The initial burn must occur at a time such that rendezvous is possible at the second burn (see Figure G.4). This launch window for a single interceptor vehicle (OSV) attempting rendezvous with a single target vehicle (satellite) occurs once every synodic period (Bate, 1971:367). The synodic period for earth orbits is defined as:

$$T_s = 2\pi / |w_1 - w_2| \quad (G.26)$$

where

T_s = Synodic period

w_1 = Angular velocity of orbit 1 relative to the center of the Earth

w_2 = Angular velocity of orbit 2 relative to the center of the Earth

But the angular velocity of an orbit is 2π divided by the orbit period. Therefore, the synodic period can be written in terms of the parking orbit and service orbit periods.

$$T_s = 1/|1/T_{po} - 1/T_{so}| \quad (G.27)$$

However, in our model every satellite is a candidate target vehicle, so the time between launch windows is:

$$\begin{aligned} t_{\text{maxphase}} &= T_s/N \\ &= 1/|N/T_{po} - N/T_{so}| \end{aligned} \quad (G.28)$$

where

N = Number of satellites in the constellation

This time represents the longest wait before next transfer initiation. It was assumed a launch opportunity had just been missed at the completion of loading the OSV. Since the OSV loading completion time is random, we chose to use the mean time between launch windows for our static model. Therefore, the average phasing time per OSV mission is one half of t_{maxphase} , or:

$$t_{\text{avgphase}} = 1/|2N/T_{po} - 2N/T_{so}| \quad (G.29)$$

Substituting $T=2\pi r^{3/2}/\sqrt{\mu}$ for each orbit period and rearranging the above equation becomes:

$$t_{\text{avgphase}} = (2\pi/\sqrt{\mu})((r_{po} * r_{so})^{3/2}/2N(r_{so}^{3/2} - r_{po}^{3/2})) \quad (G.30)$$

This equation will be used to determine the time the OSV loiters in the parking orbit. Now the transfer times will be calculated.

Since all maneuvers are impulsive the transfer time

from the parking orbit to the service orbit and back again is just the period of the transfer ellipse:

$$T_{tr} = (2\pi/\sqrt{\mu})[(r_{so}+r_{po})/2]^{3/2} \quad (G.31)$$

Notice this is Eq (G.18) with the semi-major axis a replaced by $(r_{so}+r_{po})/2$.

The time the OSV spends in the constellation can be broken into time spent servicing Y satellites and the time to travel between them. If we assume some average time to service a satellite, say Q , then the total time to service Y satellites is:

$$t_{servicing} = YQ \quad (G.32)$$

The time to travel between Y satellites is the number of trips $(Y-1)$ multiplied by the time per trip; which is simply the period of the waiting orbit. Substituting Eq (G.19), the semi-major axis of the waiting orbit, into Eq (G.18), the period of the waiting orbit, and multiplying the result by $(Y-1)$ we get:

$$t_{intersat} = (2\pi/\sqrt{\mu})r_{so}^{3/2}[(Nn-1)/N](Y-1) \quad (G.33)$$

Therefore the time to travel between and service Y satellites is:

$$t_{satserv} = (2\pi/\sqrt{\mu})r_{so}^{3/2}[(Nn-1)/N](Y-1) + YQ \quad (G.34)$$

Finally the time of flight or mission length of an OSV is the sum of Eqs (G.30), (G.31) and (G.34):

$$\begin{aligned} \text{TOF} = & (2\pi/\sqrt{\mu})[(r_{po}r_{so})^{3/2}/(2N(r_{so}^{3/2}-r_{po}^{3/2}))] \\ & + (2\pi/\sqrt{\mu})[(r_{so}+r_{po})/2]^{3/2} \\ & + (2\pi/\sqrt{u})r_{so}^{3/2}[(Nn-1)/N](Y-1) + YQ \end{aligned} \quad (\text{G.35})$$

G.8 Summary of Equations

We present here for convenience those primary equations developed in the previous sections that are used in the SSS model to represent the orbital mechanics of the OSV. See appropriate development paragraphs for explanation of terms.

Fuel Consumed by an OSV per Mission:

$$M_f = M_o\{\exp[\Delta V_{osv}/(g_e * I_{sp})] - 1\} - YP \quad (\text{G.4})$$

where

P = average delivered payload per satellite

Delta Velocity Used by OSV per Mission:

$$\Delta V_{osv} = \Delta V_{resupply} + (Y-1)\Delta V_{wo} \quad (\text{G.25})$$

where ΔV_{wo} is defined by Eq (G.24) and Eqs (G.19), (G.20), and (G.21) are used to prevent the OSV from crashing onto the Earth. $\Delta V_{resupply}$ is defined by Eq (G.11) for coplanar transfers and Eq (G.14) for non-coplanar transfers.

Time Length of an OSV Mission:

$$\begin{aligned} \text{TOF} = & (2\pi/\sqrt{\mu})[(r_{po} * r_{so})^{3/2} / (2N(r_{so}^{3/2} - r_{po}^{3/2}))] \\ & + (2\pi/\sqrt{\mu})[(r_{so} + r_{po})/2]^{3/2} \\ & + (2\pi/\sqrt{\mu})r_{so}^{3/2}[(Nn-1)/N](Y-1) + YQ \end{aligned} \quad (G.35)$$

The following table identifies those equations derived in this appendix which are used as intermediate variable equations. Also identified are the numbers of the model equations that use the intermediate variables.

Table G.2
Equation Cross-Reference

Appendix G Orbital Mechanics Equation #	Appendix F Intermediate Variable Equation #	Appendix D Associated Model Equation #
G.4	--	D.22*
G.5	1101	D.31, 32, 38, 39
G.6	137, 100	D.22, 31, 32, 38, 39
G.7	136, 110	D.31, 32, 38, 39
G.9	1109	D.31, 32, 38, 39
G.10	1108	D.31, 32, 38, 39
G.11	132	D.22
G.19	134	D.22
G.25	135	D.22
G.35	150	D.22

* The right hand side of D.22 is a rearranged version of G.4 in terms of vehicle structural mass ratio ($M_s/(M_f+M_s)$) labeled 142. This ratio is bounded to range within technologically achievable limits.

Appendix H

Computer Program VORBCS.F and Sample Output

```

C ***** DOUBLE PRECISION VERSION *****
C ***** vorbcs.f *****
C *****
C ***** THIS PROGRAM IS AN ORBITAL TRANSFER MODEL. *****
C **** original: 27 MAY 1985 ; this version: 18 JUNE 1985 ****
C *****
C ***** CAPT R LIEBER GSE-85D *****
C *****
C *****
C * THIS MODEL CALCULATES THE DELTA V AND TIME OF FLIGHT FOR *
C * AN OSV TRAVELING BETWEEN A RESUPPLY ORBIT, AND A SERVICING *
C * ORBIT DETERMINED BY THE ALTITUDE AND INCLINATION OF THE *
C * SATELLITES. IT IS INTERACTIVE AND ALLOWS THE USER TO SPEC- *
C * IFY THE ALTITUDE AND INCLINATION OF THE SATELLITE CONSTEL- *
C * LATION, THE ALTITUDE AND INCLINATION OF THE RESUPPLY ORBIT, *
C * THE TOTAL # OF SATELLITES IN THE SERVICE ORBIT, THE # OF *
C * SATELLITES TO BE SERVICED IN THE ORBIT DURING THAT MISSION, *
C * & THE MAXIMUM # OF WAITING ORBITS TO BE MODELED. THERE IS *
C * ALSO AN OPTION THAT CALCULATES PROPULSION FUEL MASS USED. *
C *
C * IF IT IS DESIRED TO CALCULATE PROPULSION FUEL MASS USED, *
C * THE USER MUST SPECIFY THE STRUCTURAL MASS OF THE OSV, THE *
C * SPECIFIC IMPULSE OF THE OSV FUEL, AND THE AVERAGE MASS *
C * OFFLOAD AT EACH SATELLITE. *
C *****
C * OUTPUT OF THIS PROGRAM IS IN THE FORM OF TWO CHARTS WRITTEN*
C * TO TWO FILES, data1 AND data2. THE CHART IN data1 GIVES THE*
C * FOLLOWING INFORMATION FOR THE # OF SATELLITES SELECTED TO *
C * BE SERVICED AND THE RANGE OF WAITING ORBITS: SEMI-MAJOR *
C * AXIS OF THE WAITING ORBIT, PERIOD OF THE WAITING ORBIT, *
C * ALTITUDE OF WAITING ORBIT AT PERIGEE IN KM ABOVE SURFACE *
C * OF EARTH, DELTA V OF EACH WAITING ORBIT, AND DELTA V OF *
C * THE RESUPPLY ORBIT. THE CHART IN data2 SHOWS THE FOLLOWING *
C * INFORMATION FOR THE # OF SATELLITES SELECTED TO BE SERVICED*
C * AND THE RANGE OF WAITING ORBITS: TOTAL DELTA V NEEDED FOR *
C * THE MISSION SPECIFIED, TIME OF FLIGHT FOR THAT MISSION AND *
C * PROPULSION FUEL MASS BURNED IN ACCOMPLISHING THE MISSION *
C * (IF FUEL MASS OPTION IS SELECTED). *
C *****
C *****
C ***** OUTPUT WRITTEN TO 'data1' and 'data2' *****
C *****
C *****
C *****
C ***** ASSUMPTIONS *****
C *****
C *****
C * 1) SATELLITES TO BE SERVICED ARE IN CIRCULAR ORBITS AT AN *
C * INCLINATION AND ALTITUDE GIVEN IN SATELLITE CONSTELLATION *
C * MODEL. *

```

```

C *****
C * 2) THE NUMBER OF SATELLITES IN EACH PLANAR ORBIT ARE *
C * EQUALLY SPACED ABOUT THAT ORBIT. THIS IS A CONSERVATIVE *
C * ESTIMATE - IF SATELLITES WERE ACTUALLY CLOSER, DELTA V *
C * REQUIRED TO SERVICE EACH SATELLITE WOULD BE MUCH LESS THAN *
C * DELTA V NEEDED IF EQUALLY SPACED. CONSEQUENTLY A PLATFORM *
C * OF SATELLITES SHOULD BE TREATED AS A SINGLE SATELLITE FOR *
C * ITS DELTA V REQUIREMENTS. *
C *****
C * 3) ALL ORBIT TRANSFERS ARE MODELED AS HOHMANN TRANSFERS. *
C * WITH TWO IMPULSIVE BURNS. IF A PLANE INCLINATION CHANGE *
C * IS INVOLVED, THE TRANSFER IS MODELED AS A HOHMANN TRANSFER *
C * WITH PLANE CHANGE (PART OF CHANGE AT BURN 1 & REST AT BURN *
C * 2). WAITING ORBIT ELLIPSES (TRANSFER ORBIT OF OSV WAITING *
C * FOR NEXT SAT) WILL BE ASSUMED TO BE INSIDE THE CIRCULAR *
C * SERVICE ORBIT, TO ALLOW A MINIMUM WAITING TIME TO BE USED. *
C * HOWEVER, THE PERIGEE OF THE WAITING ORBIT MUST BE GREATER *
C * THAN THE RADIUS OF THE EARTH + 100 NM TO NEGLECT MAJOR *
C * ATMOSPHERIC DRAG EFFECTS (AND TO AVOID RUNNING INTO THE *
C * EARTH). R(PERIGEE.WO) > REARTH + 100 NM *
C * *
C * THE CHART IN data1 FILE SHOWS THE ALTITUDE ABOVE THE EARTHS *
C * SURFACE AT PERIGEE, SO THAT YOU CAN SELECT THE NUMBER OF *
C * WAITING ORBITS NECESSARY TO KEEP PERIGEE ABOVE 100 NM. *
C *****
C * 4) SERVICING IS ASSUMED TO BE REGULARLY SCHEDULED PRE- *
C * VENTATIVE MAINTENANCE. CONSEQUENTLY EACH SATELLITE WILL *
C * BE SERVICED IN CONSECUTIVE ORDER WITH AN AVERAGE AMOUNT *
C * OF MASS OFFLOADED AT EACH SATELLITE. *
C *****
C * DELTA V REQUIREMENTS ARE DETERMINED BY THE ORBITAL *
C * MECHANICS INVOLVED. ONCE DELTA V IS OBTAINED, FUEL MASS *
C * REQUIREMENTS MAY BE OBTAINED USING THE ROCKET EQUATION: *
C *  $\Delta V = I_{sp} \cdot G \cdot \ln(\text{Mass initial} / \text{Mass final})$  *
C where Mass initial = Mass structure + Mass fuel + Mass payload *
C * *
C * and Mass final = Mass structure + Mass payload *
C * *
C *  $\Delta V \text{ TOTAL/MISSION} = (\Delta V \text{ required from rendezvous}$  *
C *  $\text{with supplies to service orbit and}$  *
C *  $\text{return}) + (\Delta V \text{ required from}$  *
C *  $\text{service orbit to waiting orbit and}$  *
C *  $\text{return}) \cdot (\text{number of satellites ser-}$  *
C *  $\text{viced} - 1)$  *
C * *
C * ONE MISSION IS DEFINED AS TIME FROM BEING SUPPLIED WITH *
C * MASS TO RETURN FOR RESUPPLY AFTER SERVICING SATELLITES. *
C * *
C * PARKING ORBIT IS DEFINED AS THE ORBIT WHERE RENDEZVOUS *
C * WITH RESUPPLIES IS ACCOMPLISHED. *
C *

```

```

C * WAITING ORBIT IS DEFINED AS THE ORBIT USED BY THE OSV *
C * WHEN WAITING FOR THE NEXT SATELLITE TO "CATCH UP" TO *
C * BE SERVICED. *
C *****
C *****
C *****
C * THE FOLLOWING RULES ARE USED FOR DOING INCLINATION CHANGES: *
C * (HOHMANN TRANSFER WITH PLANE CHANGE WITH 2 IMPULSIVE BURNS) *
C *
C * 1) IF THE INCLINATION CHANGE IS LESS THAN OR EQUAL TO ONE *
C * DEGREE. THE ENTIRE CHANGE IS ACCOMPLISHED DURING THE *
C * SECOND BURN. *
C *
C * 2) IF THE INCLINATION CHANGE IS LESS THAN OR EQUAL TO FIVE *
C * DEGREES. ONE DEGREE OF PLANE CHANGE IS DONE DURING THE *
C * FIRST BURN, AND THE REMAINDER DURING THE SECOND BURN. *
C *
C * 3) FOR PLANE CHANGES GREATER THAN FIVE DEGREES. THREE *
C * DEGREES OF PLANE CHANGE IS DONE DURING THE FIRST BURN. *
C * AND THE REMAINDER DURING THE SECOND BURN. *
C *
C * THESE RULES WERE ARRIVED AT AFTER CAREFUL COMPUTER *
C * MODELING AND ANALYSIS AND WERE FOUND TO VARY FROM THE *
C * OPTIMAL BY LESS THAN 1%. *
C *****
C *****
C *****
      double precision two,a,vso,vwoa,delvwo,vtra,vpo,vtrp,
      +tdelv,mi,fm,mf,delvp,delva,delvrs
      real mu,rearth,altsat,rsat,pi,alt,raddif,
      +altpo,i,n,altit,wmax,smax,rad,rad1,rad3,
      +osvms,isp,mpld,gc,sat,inc,i1,i2
      integer ans
      open (unit=3,file='data1')
      open (unit=4,file='data2')
      mu=3.986012E+05
c ***** MU IS GRAVITATIONAL PARAMETER OF EARTH IN KM *****
      rearth=6378.145
c ***** REARTH IS RADIUS OF EARTH IM KM *****
      write (*,*) 'what is altitude in km of satellite class'.
      +'you want to transfer to?'
      read (*,*) altsat
      write (*,*) 'what is the inclination in degrees of the',
      +'satellite class you want to transfer to?'
      read (*,*) i2
c ***** ALTSAT IS ALTITUDE OF SATELLITE CLASS IN KM *****
      write (3,*) 'PROGRAM NAME: vorbcs.f'
      write (3,*) 'EXECUTABLE FILE: vorbcs'
      write (3,*) ''
      write (3,*) 'THIS IS FILE data1. ADDITIONAL INFORMATION MAY',
      +'BE FOUND IN FILE data2.'
      write (3,*) ''
      write (4,*) 'PROGRAM NAME: vorbcs.f'

```

```

write (4,*) 'EXECUTABLE FILE: vorbcs'
write (4,*) ''
write (4,*) 'THIS IS FILE data2. ADDITIONAL INFORMATION'.
+'MAY BE FOUND IN FILE data1.'
write (4,*) ''
write (3,*) 'DOUBLE PRECISION MODEL'
write (4,*) 'DOUBLE PRECISION MODEL'
write (*,*) 'ALTITUDE OF SATELLITE CLASS IS ',altsat,'KM'.
+'AND INCLINATION OF CLASS IS ',i2,'DEGREES'
write (3,*) 'ALTITUDE OF SATELLITE CLASS IS ',altsat,'KM'.
+'AND INCLINATION OF CLASS IS ',i2,'DEGREES'
write (4,*) 'ALTITUDE OF SATELLITE CLASS IS ',altsat,'KM'.
+'AND INCLINATION OF CLASS IS ',i2,'DEGREES'
rsat=altsat+rearth
write(*,*) ''
write (*,*) 'what is altitude in km of parking orbit for'.
+'rendezvous with resupplies?'.
+'(100nm=185.2,150nm=277.8,200nm=370.4)'
C*ALTPO IS ALTITUDE OF PARKING ORBIT FOR RZ WITH RESUPPLIES IN KM
read (*,*) altpo
write(*,*) 'what is the inclination in degrees of the'.
+'parking orbit?'
read(*,*) il
write (*,*) 'ALTITUDE OF PARKING ORBIT IS ',altpo,' KM'.
+'AND INCLINATION IS ',il,'DEGREES'
write (3,*) 'ALTITUDE OF PARKING ORBIT IS ',altpo,' KM'.
+'AND INCLINATION IS ',il,'DEGREES'
write (4,*) 'ALTITUDE OF PARKING ORBIT IS ',altpo,' KM'.
+'AND INCLINATION IS ',il,'DEGREES'
alt=altpo+rearth
inc=abs(i2-il)
pi=3.1415927
write (*,*) ''
write (*,*) 'enter the # of satellites in the orbit'.
+'to which you are transferring'
read (*,*) smax
write (*,*) '# OF SATELLITES IN ORBIT IS ',smax
write(*,*) ''
write (*,*) 'enter max # of orbits you want modeled.'
+'for waiting orbit'
write (*,*) ''
write (*,*) 'NOTE: # OF WAITING ORBITS IS THE NUMBER'.
+'OF REVOLUTIONS OSV MAKES IN TRAVELING FROM SATELLITE TO'.
+'SATELLITE. FOR MORE WAITING ORBITS. LESS DELTA V AND'.
+'PROPULSION FUEL IS USED. BUT MORE TIME IS REQUIRED.'
read (*,*) wmax
write (*,*) 'MAX # OF ORBITS MODELED IN WAITING ORBIT'.
+'WILL BE ',wmax
write(*,*) ''
write (*,*) 'how many satellites do you want to service'.
+'on one mission?'
read(*,*) sat
write (*,*) ''

```

```

write(*,*)'THIS PROGRAM WILL CALCULATE TOTAL DELTA V.',
+'TIME OF FLIGHT, AND (IF DESIRED) PROPULSION FUEL MASS',
+'NECESSARY FOR A MISSION SERVICING',sat,'SATELLITES OUT',
+'OF A TOTAL CONSTELLATION OF',smax,'SATELLITES'
write(*,*) ''
write (3,*) ''
write(3,*)'THIS PROGRAM WILL CALCULATE TOTAL DELTA V.',
+'TIME OF FLIGHT, AND (IF DESIRED) PROPULSION FUEL MASS',
+'NECESSARY FOR A MISSION SERVICING',sat,'SATELLITES OUT',
+'OF A TOTAL CONSTELLATION OF',smax,'SATELLITES'
write(3,*) ''
write (4,*) ''
write(4,*)'THIS PROGRAM WILL CALCULATE TOTAL DELTA V.',
+'TIME OF FLIGHT, AND (IF DESIRED) PROPULSION FUEL MASS',
+'NECESSARY FOR A MISSION SERVICING',sat,'SATELLITES OUT',
+'OF A TOTAL CONSTELLATION OF',smax,'SATELLITES'
write(4,*) ''
write(*,*)'do you want to calculate osv propulsion fuel',
+'burned? answer with number: yes=1 or no=0'
read (*,*) ans
if (ans.eq.0) go to 400
write(*,*) ''
write (*,*) 'What is the empty structural mass of your',
+'OSV in kg?'
read (*,*) osvms
write (*,*) 'OSV STRUCTURAL MASS IS',osvms,'KG'
write (3,*) 'OSV STRUCTURAL MASS IS',osvms,'KG'
write (4,*) 'OSV STRUCTURAL MASS IS',osvms,'KG'
write(*,*) ''
write (*,*) 'What is the specific impulse of the OSV',
+'fuel in sec?'
read (*,*) isp
write (*,*) 'OSV FUEL Isp =',isp,'SEC'
write (3,*) 'OSV FUEL Isp =',isp,'SEC'
write (4,*) 'OSV FUEL Isp =',isp,'SEC'
write(*,*) ''
write (*,*) 'What is the average mass offload at a',
+'satellite in kg?'
read (*,*) mpld
write (*,*) 'AVE PAYLOAD OFFLOAD AT EACH SATELLITE =',
+'mpld,'KG'
write (3,*) 'AVE PAYLOAD OFFLOAD AT EACH SATELLITE =',
+'mpld,'KG'
write (4,*) 'AVE PAYLOAD OFFLOAD AT EACH SATELLITE =',
+'mpld,'KG'
write(*,*)"PLEASE STANDBY - I'M BUSY COUNTING!!"
C **** gc is acceleration of gravity *****
gc=9.807E-03
write(3,*)'FUEL MASS IS TOTAL OSV FUEL USED FOR PROPULSION',
+'IN KG'
write(4,*)'FUEL MASS IS TOTAL OSV FUEL USED FOR PROPULSION',
+'IN KG'
400 write(3,*)'PERIOD OF THE WAITING ORBIT IS IN HOURS'

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AD-A172 478

A METHODOLOGY FOR SELECTION OF A SATELLITE SERVICING

2/2

ARCHITECTURE VOLUME 3 (U) AIR FORCE INST OF TECH

WRIGHT-PATTERSON AFB OH SCHOOL OF ENGI..

UNCLASSIFIED

J W ANDERSON ET AL. DEC 85

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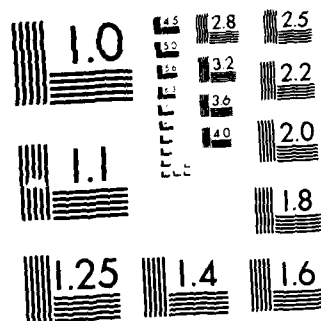
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XEROCOPY RESOLUTION TEST CHART
NATIONAL BUREAU OF STANDARDS-1963-A

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write(4,*)'PERIOD OF THE WAITING ORBIT IS IN HOURS'
write(3,*)''
write(3,*)'THE # OF WAITING ORBITS INDICATES THE NUMBER OF',
+'REVOLUTIONS REQUIRED TO TRANSIT FROM ONE SATELLITE TO THE',
+'NEXT NOTE THAT THE MORE WAITING ORBITS USED BETWEEN',
+'SATELLITES. THE LESS DELTA V AND PROPULSION FUEL NEEDED.'
write(3,*)''
write(4,*)''
write(4,*)'THE # OF WAITING ORBITS INDICATES THE NUMBER OF',
+'REVOLUTIONS REQUIRED TO TRANSIT FROM ONE SATELLITE TO THE',
+'NEXT NOTE THAT THE MORE WAITING ORBITS USED BETWEEN',
+'SATELLITES. THE LESS DELTA V AND PROPULSION FUEL NEEDED.'
write(4,*)''
write(*,*)''
write(*,*) "PLEASE STANDBY - I'M BUSY COUNTING!!"
write(3,*) 'DELTA V IS IN KM/HR'
write(3,*) ''
write(4,*) 'DELTA V IS IN KM/HR'
write(4,*) ''
write(3,*)'*****'
write(4,*)'*****'
write(3,*)''
write(4,*)''
write(3,10)
10 format('# OF SATS'.4X,'# OF SATS'.4X,'# OF WAITING ORBITS',
+6X,'PERIOD.WO',10X,'A.WO',10X,'WO ALT ABV'.8X,'DELTA V.WO',
+5X,'DELTA V.RESUP')
write(3,11)
11 format(1X,'SERVICED'.4X,'IN PLANE'.33X,'HRS'.14X,'KM'.13X,
+'EARTH. KM',10X,'KM/HR',11X,'KM/HR')
write(3,*) ''
IF (ans.eq.0) go to 20
write(4,25)
25 format ('# OF SATS'.4X,'# OF SATS'.4X,'# OF WAITING ORBITS',
+4X,'TOTAL DELTA V'.4X,'TIME OF FLIGHT'.4X,'FUEL MASS')
write(4,12)
12 format(1X,'SERVICED'.4X,'IN PLANE'.31X,'KM/HR'.13X,'HRS',
+12X,'KG')
write(3,*) ''
go to 30
20 write(4,26)
26 format ('# OF SATS'.4X,'# OF SATS'.4X,'# OF WAITING ORBITS',
+4X,'TOTAL DELTA V'.4X,'TIME OF FLIGHT')
write(4,13)
13 format(1X,'SERVICED'.4X,'IN PLANE'.31X,'KM/HR'.13X,'HRS')
write(3,*) ''
30 do 100 n=smax,smax
C ***** N IS NUMBER OF SATELLITES PER ORBIT *****
do 200 i=1,wmax
C ***** I IS NUMBER OF ORBITS IN WAITING ORBITS *****
two=((n*i-1)/(n*i))*(rsat**(3./2.))*2.*pi/sqrt(mu)
C ***** TWO IS PERIOD OF WAITING ORBIT *****
ttro=2.*pi*sqrt((((alt+rsat)/2.)*3.)/mu)

```

```

C ***** TTRO IS PERIOD OF TRANSFER ORBIT (PO TO SO) *****
      tof=ttro+two*(sat-1.)
C ***** TOF IS TIME OF FLIGHT FOR ONE COMPLETE MISSION *****
      a=(two*sqrt(mu)/(2.*pi))*(2./3.)
C ***** A IS SEMI-MAJOR AXIS OF WAITING ORBIT *****
      altit=2.*a-rsat-rearth
C ***** ALTIT IS ALTITUDE ABOVE EARTH AT PERIGEE OF WO *****
      vso=sqrt(mu/rsat)
C ***** VSO IS SPEED IN SERVICE ORBIT *****
      vwoa=sqrt(2.*(mu/rsat-mu/(2.*a)))
C ***** VWOA IS SPEED IN WAITING ORBIT AT APOGEE *****
      delvwo=2.*dabs(vso-vwoa)
C ** DELVWO IS DELTA V FOR TRAVELING BETWEEN SATS (WAITING ORBIT)
      vtra=sqrt(2.*abs(mu/rsat-mu/(alt+rsat)))
C **** VTRA IS SPEED IN TRANSFER ORBIT AT APOGEE FROM SO TO PO **
      vpo=sqrt(mu/alt)
C ***** VPO IS VELOCITY OF CIRCULAR PARKING ORBIT *****
      vtrp=sqrt(2.*abs(mu/alt-mu/(alt+rsat)))
C * VTRP IS SPEED OF OF TRANSFER ORBIT AT PERIGEE FROM PO TO SO *
C
C *****
C * THE FOLLOWING RULES ARE USED FOR DOING INCLINATION CHANGES: *
C *
C * 1) IF THE INCLINATION CHANGE IS LESS THAN OR EQUAL TO ONE *
C *    DEGREE, THE ENTIRE CHANGE IS ACCOMPLISHED DURING THE *
C *    SECOND BURN. *
C *
C * 2) IF THE INCLINATION CHANGE IS LESS THAN OR EQUAL TO FIVE *
C *    DEGREES, ONE DEGREE OF PLANE CHANGE IS DONE DURING THE *
C *    FIRST BURN, AND THE REMAINDER DURING THE SECOND BURN. *
C *
C * 3) FOR PLANE CHANGES GREATER THAN FIVE DEGREES, THREE *
C *    DEGREES OF PLANE CHANGE IS DONE DURING THE FIRST BURN, *
C *    AND THE REMAINDER DURING THE SECOND BURN. *
C *
C * THESE RULES WERE ARRIVED AT AFTER CAREFUL COMPUTER *
C *    MODELING AND ANALYSIS AND WERE FOUND TO VARY FROM THE *
C *    OPTIMAL BY LESS THAN 1%. *
C *****
C *
C * RAD IS THE PLANE INCLINATION DIFFERENCE IN RADIANS *
      rad=inc*pi/180.
C * RAD1 IS ONE DEGREE IN RADIANS ****
      rad1=pi/180.
C * RAD3 IS THREE DEGREES IN RADIANS ****
      rad3=pi/60.
      IF (inc.le.1.0) THEN
        delvp=dsqrt((vpo**2.0)+(vtrp**2.0)-2.0*vpo*vtrp)
        delva=dsqrt((vso**2.0)+(vtra**2.0)-2.0*vso*vtra*cos(rad))
      ELSE IF (inc.le.5.0) THEN
        delvp=dsqrt((vpo**2.0)+(vtrp**2.0)-2.0*vpo*vtrp*cos(rad1))
        raddif=rad-rad1
        delva=dsqrt((vso**2.0)+(vtra**2.0)-2.0*vso*vtra*cos(raddif))

```

```

ELSE
delvp=dsqrt((vpo**2.0)+(vtrp**2.0)-2.0*vpo*vtrp*cos(rad3))
raddif=rad-rad3
delva=dsqrt((vso**2.0)+(vtra**2.0)-2.0*vso*vtra*cos(raddif))
ENDIF
delvrs=2.*(delvp+delva)
C ***** DELVRS IS DELTA V FOR RESUPPLY *****
tdelv=delvwo*(sat-1)+delvrs
C ** TDELV IS TOTAL DELTA V **
if (ans.eq.0) go to 70
C ** mi IS MASS INITIAL, mf IS MASS FINAL, fm IS FUEL MASS *****
mi=osvms*dexp(delvrs/(2.*isp*gc))
fm=mi-osvms
mf=mi+mpld
IF (sat.eq.1.) go to 300
do 300 j=1,sat-1.
mi=mf*dexp(delvwo/(isp*gc))
fm=fm+mi-mf
mf=mi+mpld
300 continue
mi=mf*dexp(delvrs/(2.*isp*gc))
fm=fm+mi-mf
C ***** CHANGE TIME IN ANSWERS FROM SEC TO HOURS *****
70 two=two/3600.
delvwo=delvwo*3600.
delvrs=delvrs*3600.
tdelv=tdelv*3600.
tof=tof/3600.
write (3,50) sat,n,i,two,a,altit,delvwo,delvrs
50 format (F5.0,8x,F5.0,15x,F5.0,7x,5D17.9)
IF (ans.eq.0) go to 60
write (4,75) sat,n,i,tdelv,tof,fm
75 format (F5.0,8x,F5.0,15x,F5.0,8x,3D17.9)
go to 200
60 write (4,76) sat,n,i,tdelv,tof
76 format (F5.0,8x,F5.0,15x,F5.0,8x,2D17.9)
200 continue
100 continue
write (*,*) "INFORMATION IS IN FILE 'data1' and 'data2'"
write (3,*) ''
write (4,*) ''
write (3,*) '*****'
write (4,*) '*****'
end

```

PROGRAM NAME: vorbes.f
EXECUTABLE FILE: vorbes

THIS IS FILE data1. ADDITIONAL INFORMATION MAY BE FOUND IN FILE data2.

DOUBLE PRECISION MODEL

ALTITUDE OF SATELLITE CLASS IS 800.000 KM AND INCLINATION OF CLASS IS 0. DEGREES

ALTITUDE OF PARKING ORBIT IS 185.200 KM AND INCLINATION IS 0. DEGREES

THIS PROGRAM WILL CALCULATE TOTAL DELTA V,
TIME OF FLIGHT, AND (IF DESIRED) PROPULSION FUEL MASS
NECESSARY FOR A MISSION SERVICING 5.00000 SATELLITES OUT
OF A TOTAL CONSTELLATION OF 144.000 SATELLITES

OSV STRUCTURAL MASS IS 2000.00 KG

OSV FUEL Isp = 450.000 SEC

AVE PAYLOAD OFFLOAD AT EACH SATELLITE = 250.000 KG

FUEL MASS IS TOTAL OSV FUEL USED FOR PROPULSION IN KG

PERIOD OF THE WAITING ORBIT IS IN HOURS

THE # OF WAITING ORBITS INDICATES THE NUMBER OF
REVOLUTIONS REQUIRED TO TRANSIT FROM ONE SATELLITE TO THE
NEXT NOTE THAT THE MORE WAITING ORBITS USED BETWEEN
SATELLITES, THE LESS DELTA V AND PROPULSION FUEL NEEDED.

DELTA V IS IN KM/HR

# OF SATS SERVICED	# OF SATS IN PLANE	# OF WAITING ORBITS	PERIOD,WO HRS	A.WO KM	WO ALT ABV EARTH, KM
5.	144.	1.	0.166955200d+01	0.714487395d+04	0.733457886d+03

11/2

WO ALT ABV
EARTH, KM

DELTA V, WO
KM/HR

DELTA V, RESUP
KM/HR

733457885d+03 0.125065556d+03 0.245542030d+04

292

PROGRAM NAME: vorbcs.f
EXECUTABLE FILE: vorbcs

THIS IS FILE data2. ADDITIONAL INFORMATION MAY BE FOUND IN FILE data1.

DOUBLE PRECISION MODEL

ALTITUDE OF SATELLITE CLASS IS 800.000 KM AND INCLINATION OF CLASS IS 0. DEGREES

ALTITUDE OF PARKING ORBIT IS 185.200 KM AND INCLINATION IS 0. DEGREES

THIS PROGRAM WILL CALCULATE TOTAL DELTA V,
TIME OF FLIGHT, AND (IF DESIRED) PROPULSION FUEL MASS
NECESSARY FOR A MISSION SERVICING 5.00000 SATELLITES OUT
OF A TOTAL CONSTELLATION OF 144.000 SATELLITES

OSV STRUCTURAL MASS IS 2000.00 KG

OSV FUEL Isp = 450.000 SEC

AVE PAYLOAD OFFLOAD AT EACH SATELLITE = 250.000 KG

FUEL MASS IS TOTAL OSV FUEL USED FOR PROPULSION IN KG

PERIOD OF THE WAITING ORBIT IS IN HOURS

THE # OF WAITING ORBITS INDICATES THE NUMBER OF
REVOLUTIONS REQUIRED TO TRANSIT FROM ONE SATELLITE TO THE
NEXT NOTE THAT THE MORE WAITING ORBITS USED BETWEEN
SATELLITES, THE LESS DELTA V AND PROPULSION FUEL NEEDED.

DELTA V IS IN KM/HR

# OF SATS SERVICED	# OF SATS IN PLANE	# OF WAITING ORBITS	TOTAL DELTA V KM/HR	TIME OF FLIGHT HRS	FUEL MASS KG
5.	144.	1.	0.295568253d+04	0.825260353d+01	0.530880867d+03

15/2

0.

5

WIGHT FUEL MASS
KG
sd-01 0.530880867d+03

2.32

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Captain Delbert B. Langerock was born on 16 November 1954 in El Paso, Texas. He graduated from high school in Marion, South Dakota in 1973 and entered the Air Force as a Telecommunications Control Specialist. Following tours of duty at RAF Uxbridge (Hillingdon), England and NORAD Combat Operations Center, Colorado, he was selected into the Airmen Education and Commissioning Program in 1978. He attended Iowa State University from which he received the degree of Bachelor of Science in Industrial Engineering in May 1981. Upon graduation, he attended USAF Officer Training School and received a commission in the USAF in August 1981. He then served as a Minuteman II System Technical Engineer in the 44th Strategic Missile Wing, Ellsworth AFB, South Dakota, until entering the School of Engineering, Air Force Institute of Technology, in May 1984.

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VOLUME III - APPENDICES

A two-phase methodology for selecting an optimal military satellite servicing system is developed using the systems engineering approach. This methodology is used to evaluate several alternative systems at varying levels of detail. The candidate systems are composed of low-G launchers, high-G launchers, orbital servicing vehicles, and space bases. An optimal realization is then derived for a system of low-G launchers and orbital servicing vehicles. In the first phase of the approach, vector optimization techniques are used to vary the states of a model to obtain a set of optimal solutions. The second phase embodies the decision maker's preferences in a value system to enable preference ranking of the optimal solutions in the non-dominated solution set. This methodology, as presented, can be applied to any complex problem with multiple conflicting objectives. It is designed for use by an engineering organization supporting a senior-level decision maker.

The report is in three volumes. The Executive Summary (Volume I) is a cursory review of the study and is meant to be self-contained. The Final Report (Volume II) and the Appendices (Volume III) are more detailed and should be read together for completeness.

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